AD705370

AGARD CP No. 62

AGARD CONFERENCE PROCEEDINGS No. 62

AGARD

ADVISORY GROUP FOR AEROSPACE RESEARCH & DEVELOPMENT

7 RUE ANCELLE 92 NEUILLY SUR SEINE FRANCE

Preliminary Design Aspects of Military Aircraft

NORTH ATLANTIC TREATY ORGANIZATION

DISTRIBUTION AND AVAILABILITY ON BACK COVER

281

AGARD Conference Proceedings No.62

NORTH ATLANTIC TREATY ORGANIZATION

ADVISORY GROUP FOR AEROSPACE RESEARCH AND DEVELOPMENT

(ORGANISATION DU TRAITE DE L'ATLANTIQUE NORD)

PRELIMINARY DESIGN ASPECTS OF MILITARY AIRCRAFT

A selection of papers presented at the 35th Meeting of the Flight Mechanics Panel of AGARD
held at The Hague, The Netherlands, 2-5 September 1969

Published March 1970

623.746

Printed by Technical Editing and Reproduction Ltd
Harford House, 7-9 Charlotte St. London. W1P 1HD

<u>N O T E</u>

The papers published in this volume do not represent the complete proceedings of the AGARD Conference on "Preliminary Design Aspects of Military Aircraft" held in The Hague from 2 to 5 September 1969. The meeting was classified "NATO SECRET", and, therefore, 6 classified papers, the abstract of the round table discussion and the discussion forms will be published by AGARD in a second part of the conference proceedings and will be sent to the Ministry of Defense at each NATO country. If you are interested in this classified part, please contact the MOD or the AGARD National Distribution Centre of your country.

Col. Ludwig WÖLKER
Executive of the AGARD
Flight Mechanics Panel

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on

PRELIMINARY AIRPLANE DESIGN

by

Clem C. WEISSMAN, The Pentagon, U.S.A.

Introductory Remarks

for the

35th Meeting of the FMP

on the subject of

PRELIMINARY AIRPLANE DESIGN

by Mr. Clem C. Weissman

Ladies and Gentlemen:

I quote from one of the most recent AGARD bulletins :

"The Advisory Group for Aerospace Research and Development (AGARD) is a NATO agency under the authority of the Military Committee." The following vugraph shows the mission of AGARD as given in this bulletin.

Mission (Vugraph 1)

As it may have been said by the philosopher, Heraclitus, in about 500 B.C., "All things are in process and nothing stays still", so is it with AGARD and NATO. Not only has the organization of NATO changed; not only has the word "aeronautical" changed to "aerospace"; but, the mission, as seen here, has also undergone some change from the objectives as set forth by Dr. Theodore Von Karman and the first national delegates meeting in February 1951. The objectives at that time in Von Karman's writings included the following words :

1. "To review continuously the application of advances in aeronautical science to common defense problems" and

2. "To make recommendations for the solution of problems referred to it by agencies within NATO, including evaluation of research and development projects submitted by individual nations."

To some of us, the trend in AGARD, as it has grown since 1952 when it consisted mainly of the National Delegates to todays' Panel and Committee structure involving approximately 350 panel members; has been highly specialized and away from the "application of advances" to a complete airplane design that may solve a NATO problem.

The Flight Mechanics Panel in sponsoring this meeting today desires to highlight and explore a problem that involves an evaluation of the specialized research and development projects carried on in individual nations. That is, the designer's problem, the task of achieving a correct balance between the different disciplines, aerodynamics, propulsion, structures, avionics and others involved in producing a total airplane system to meet a military need. As stated elsewhere before, the object of this symposium is to bring together the designer and the specialist, to highlight the methods and techniques of preliminary airplane design and to explore those compromises involved in the early design stage that so often frustrate the dedicated specialist.

It is the hope of the program coordinators that this meeting will also serve the additional purpose of improving the cooperation among us. The presentations that follow these opening remarks will illustrate different approaches to design, different engineering and management practices and different national values placed on the manager, engineer and scientist. The program coordinators encourage you to keep in mind the differences in airplanes of the same type when one is able to develop a prototype rather than a production model first.

With the present complex state of aircraft technology it is no longer possible for the designer to "invent" his own aircraft and get them developed through to the flight stage. The expense involved in the development process leads to the necessity for major external funding, usually in the case of military aircraft from government sources or possibly from groups of governments. This in turn has led to major external vetting of all stages of the design process.

The stages usually found in a development program are shown in Figure 1. The names given to these stages may differ from country to country, but, the essential actions are the same. The modern military design starts with a draft requirement set out by the military authorities for an aircraft to meet an anticipated military need. In this context, we welcome the S.H.A.P.E. representatives here today, and in the seventh session we will hear from them their views on the direction in which the requirements for military combat aircraft are likely to go in the future.

At this initial stage the requirement is set out in generalized terms. Feasibility studies are undertaken to this requirement to outline the various possible technical solutions, together with indications of time-scale and cost. This enables an initial study of the relative cost effectiveness of alternative proposals to be determined (Figure 2). From the knowledge obtained from the feasibility studies as to what is technically possible, the military authorities can produce a firm requirement. One or more project studies will then be carried out against the full requirement. These project studies explore in depth an aircraft proposal and end up with a complete aircraft specification with detailed performance, development and production time scales and costs. It is during this phase that all the major technical problems must be investigated and solved, and schemes must be completed for the whole design (Figure 3).

When the specification is accepted and a design go-ahead is given, the major compromises and decisions have already been made and only detailed design and manufacture of the first aircraft remains. The necessity for these decisions to be made during the project study stage is often not clearly understood by those responsible for the requirements and the funding of military aircraft. Excessive delays and major cost increases are incurred when changes to a requirement are made after a design go-ahead has been given.

In the preliminary design phase - during feasibility and project studies - all facets of technology have to be considered. During this symposium we will be reviewing these various disciplines with sessions on Aerodynamics, Power Plant, Structures, Airframe Systems and Operational Systems. In each session there will be an introductory paper reviewing the design problem, followed by specialist papers.

The Flight Mechanics Panel is grateful for the help they have received from the Fluid Dynamics Panel, the Propulsion and Energetics Panel, the Structures and Materials Panel, the Guidance and Control Panel and the Avionics Panel in arranging these specialist papers.

In the past ten years, there has been a serious deficiency of new design aircraft in flight test in the United States. Comparatively speaking there are many new design innovations in flight status in the European countries. The advancement of state-of-the-art in aviation requires all the stages from research through exploratory development, engineering development and flight test to production of a total aircraft system. It is hoped that this meeting contributes to that advancement by introducing in AGARD the subject of the preliminary design aspects of military aircraft.

Vugraph 1

MISSION

The mission of AGARD is to bring together the leading personalities of the NATO nations in the fields of science and technology relating to aerospace for the following purposes :

a. Recommending effective ways for the member nations to use their research and development capabilities for the common benefit of the NATO community;

b. Providing scientific and technical advice and assistance to the Military Committee in the field of aerospace research and development ;

c. Continuously stimulating advances in the aerospace sciences relevant to strengthening the common defence posture;

d. Improving the co-operation among member nations in aerospace research and development;

e. Exchanging of scientific/technical information;

f. Providing assistance to member nations for the purpose of increasing their scientific and technical potential;

g. Rendering scientific and technical assistance, as requested, to other NATO bodies and to member nations in connection with research and development problems in the aerospace field.

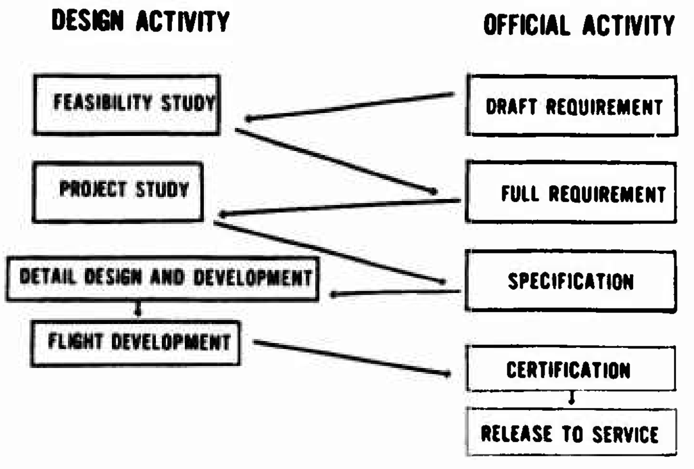

Fig.1 Development Programme

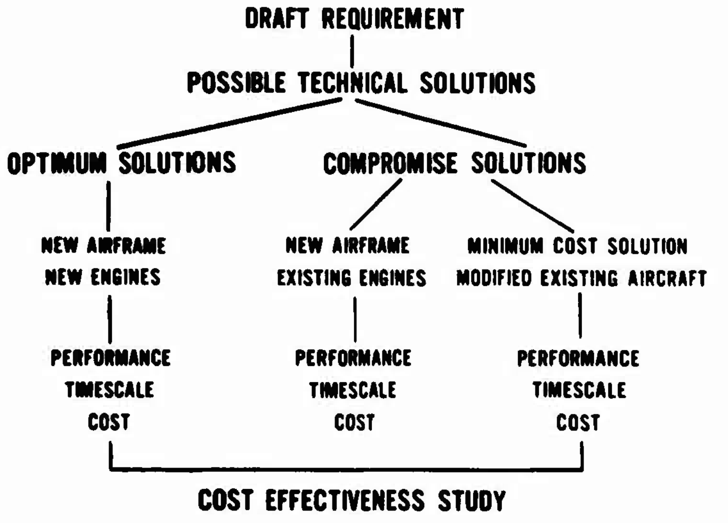

Fig.2 Feasibility Study

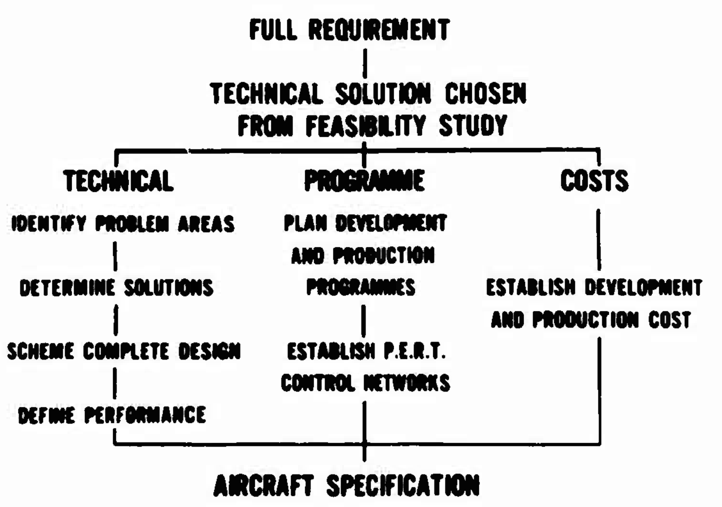

Fig.3 Project Study

PROJECT DESIGN OF COMBAT AIRCRAFT

by

B.O.Heath

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SUMMARY

This paper is intended to serve as an introduction to the later specialist sessions by showing how project design must increasingly call upon the many specialist skills in a modern aircraft organisation embracing design, test, production and commercial directorates, in association with engine, equipment and avionic firms: attention to absolute cost in addition to cost effectiveness has promoted new co-operative functions. Project Design has become a process for anticipating with confidence the type of complete declaration of performance, integrity, timescale and costs previously often only emerging as detail stages were completed: the traditional role of establishing a datum aircraft configuration is covered as one part of a project design process which as a whole must ensure margins and tolerances acceptable to both contractor and procurement agency.

PROJECT DESIGN OF COMBAT AIRCRAFT

B.O. Heath

1. INTRODUCTION

Some aspects of project design activities are presented against a United Kingdom background, which has become increasingly influenced by international collaboration since 1965.

Up to the 1950's it was a common expectation in Great Britain and elsewhere to progressively develop aircraft as members of a growing family, each 'Mark' giving either an improvement in performance - typically range or speed - or in armament and equipment fit specified to meet any particular new role. In many cases at least the initial phases of development were insured by the Government sponsoring parallel activities on prototypes within two or even three firms, sometimes with small scale versions preceding the main aircraft where innovation (often size) was considered sufficiently great to warrant this. The V-Bombers of the 1950's give one example of this latter aspect.

Although long production life and versatile application still remain the final aims, there are many changes in the way this now has to be secured: after competitive phases a fighter strike aircraft is developed by a single chosen firm, the first aircraft having a first flight date several years after that of its predecessor; in the United Kingdom true prototypes are not usual, Service clearance being attained through a 'Development Batch' of aircraft with a standard expected closely to resemble that for the Service aircraft. These have to give a high specified performance from the start, embracing a wide variety of requirements foreseen as covering the needs of many years to come.

There were many reasons for this change in procedure: within U.K. it was directly sponsored by requirements from the Zuchemann plan which itself reflected deeper reasons such as led to the industrial rationalisation. These included the desire to avoid duplication, reaction to the difficulties from trying to extrapolate subsonic development across the speed of sound which had led to some expensive but finally abortive propositions; the higher development content and greater absolute price of the subsequent aircraft which did provide genuine supersonic capabilities. The customer had a greater wish to be convinced as to the operational adequacy of any proposal: he required greater 'visibility' throughout and, with the contractor, wished to avoid expensive and frustrating modifications by ensuring the best match between operational requirements and what was feasible technically, respecting reasonable cost and for a given Service Programme.

All this has changed and increased the scope of work in what was previously considered initial design. The length of time between completely new designs has introduced a much greater element of decision as to the degree of innovation to embody, balancing potential risk against rewards. This has to be justified to the customer and his technical agencies who will be open to the claims and attractions of a variety of existing aircraft some of which will have had the benefit of flight development and have thereby a known, even if somewhat inferior capability. The merits of such rival aircraft, particularly in aggregate, provide a great challenge to the project design team to provide aircraft with sufficient advantage in capability and economy as to justify commencement of his own project. He must not be tempted to promise too much.

Such considerations, and the fact that economic and operational needs set most fighter-strike aircraft on the Design Mach Number plateau around M = 2 (essentially light alloy structure subject to progressive but containable kinetic heating, directional stability and stiffness requirements) mean that the better elements of the prototype plus progressive development approach again have attractions in establishing and exploiting basic capabilities by the earliest flight development.

2. ORGANISATIONAL ASPECTS

The time between major new designs at least has the advantage - even in a multi-project organisation - that a balanced design and estimating team is available from the start of a new project. A share of wind tunnel and computer time is available and procedures and programmes developed on other designs make it possible to produce 'point designs' quite rapidly. One of these is likely to receive more high quality technical input than did a mainstream design of the 1950's. Cost estimators, production planners and Service Department engineers form part of the project design team: much progress has been made using data compiled from earlier projects and under the stimulus of formal Value Engineering to provide background cost data similar to that available in structural and aerodynamic activities, so that reliable absolute costs may be derived, and optimum solutions pursued as far as is sensible. For example, the true cost of processes which a few years ago were covered by general rates are now known. Progress has been remarkable: a few years ago Engineers were not allowed to see cost submissions: now they are increasingly responsible for detail contributions to them and for maintaining the close budgets which result.

The means whereby overall integration of effort is secured within one whole organisation is briefly described.

B.A.C. Preston Division use an organisation of the project management/line management matrix type wherein line management are responsible for the technical or specialist adequacy of the aircraft and for the detailed implementation of requirements. They feed into each project the experience from others and are responsible for maintaining their cost and timescale commitments against estimates agreed with project management. The Project Manager is responsible for ensuring compatibility across the line directorates and ensuring that customer requirements of cost, performance and timescale are interpreted, promulgated and actioned.

A senior project representative with allegiance to both project and line management is established at each node of the matrix. During the feasibility and definition phases such personnel are detached from the main organisation to represent their Department or specialist views in the overall synthesis of the project proposals within the Project Office. When the project expands in later phases they return to their departments giving continuity and providing a local nucleus for manpower expansion.

The project office has permanent overall planning, monitoring and control groups and nominees for overall cost control in very close co-operation with the Estimating/Financial/Commercial complex on task definition, estimating, monitoring and control.

Apart from final contractual and financial aspects the Project Manager provides the official contact with the customer and in many ways represents him within the firm. The establishing of an analogous Project Team within the customer's agency has proved to be a great advantage, allowing the development of quick reacting machinery for reaching decisions, economic reporting, and fruitful, frank discussion of problems and their solution making the maximum use of all resources.

3. PARAMETRIC STUDIES

Parametric analysis has not been excluded by the capability for rapid synthesis of point designs in some detail: such analysis is used to establish first-order areas of design solutions as part of establishing a datum (Fig.1), to investigate rates of exchange between performance parameters and between performance and weight. Growth factors for assessing the overall effect of extra specific items and if existing engines are not suitable, the sizing of the engine and matching of its thermodynamic cycle to the needs of the requirement often appeal to parametric models.

However, aircraft definition is not the continuous process often postulated by parametric studies, and the earliest opportunity is taken to ensure by a physical study and point designs that some important but possibly peculiar non-uniformity does not in fact invalidate a general trend. Some ingenious layout may apply on one side of a given sizing since elements of an aircraft scale at different rates or conversely a design may integrate well at one size but not when much removed from it.

Throughout the design process there is a need for immediate and concise guidance for engineers and designers so that their decisions can be as soundly based as possible and ascribable to a given source. However, in contrast to the later stages of development when for example Design Office Standard and Stressing Data Sheets are precise and stable, there are many areas in the early stages of design where corresponding precision is not possible and where a clear presentation is difficult even if a correlation exists because of the large number of variables involved. Initial weight estimation of swept wings was once shown to be given as reliably by a simple percentage of gross weight as by complex formulae connecting aircraft weight, wing loading, design speed, sweep, thickness, taper and aspect ratio.

Taking wing weight as an example, the earlier availability of stress calculations for main structure remains somewhat vulnerable as far as weight is concerned because of the comparatively large contribution of secondary structure so that parametric methods remain very useful. One way of resolving this dilemma is to plot component weights against some very simple function such as structural overhang ratio, not expecting to obtain correlation, but to use judgement in selecting relevant points from the cloud obtained. Figure 2 shows such a cloud from many wings. This may then prompt correlating factors which are powerful enough to secure good correlation. Figure 3 shows an abstract for delta wings from Figure 2 in which meaningful strength and stiffness parameters (with a correction for design temperature) gave excellent correlation by separating out those delta wings which were designed by one or other parameter. Normally this legitimate engineering difference would have appeared as a scatter. The wing when built showed the validity of the approach.

3.1 Initial Sizing

The forces which act on an aircraft (lift, drag, thrust and weight/inertia) and the areas which support them (wing and intake area, wetted area, cross sectional and base areas) provide the main overall parameters requiring definition. They are conveniently grouped into ratios such as

$$\frac{T}{W},\ \frac{W}{S},\ \frac{L}{D}$$

$$\frac{\text{Wetted}}{\text{Ref. Area}}$$

$$\frac{\text{Payload}}{\text{Total Wt.}}$$

$$\frac{\text{Fuel}}{\text{Total Weight}}$$

and, given certain supporting data, it is possible to express most performance parameters in terms of these and (say) total weight.

For example, the boundaries of sustained capability of an aircraft embracing maximum speed, ceiling, either at 1'g' or when manoeuvering are all obtained when specific excess power

$$S.E.P. = (T-D)\frac{V}{W}$$

is equal to zero. S.E.P. is therefore regarded as one measure of combat capability and it is now usual for at least one value of it to be specified at operationally meaningful conditions as a datum requirement it is equivalent to steady rate of climb.

Again, most sorties are made up from individual elements of cruise over which the Breguet Range Equation applies

$$\text{Range} = \frac{V}{sfc} \times \frac{L}{D} \log \frac{W_2}{W_1}$$

Further, for so-called transient manoeuvring i.e. forces balanced normal to the flight path but not necessarily along it

$$L = (n\,W - P) = C_L\, q\, S$$

(a case which covers landing and take-off).

Take-off distances are given by an equation of the form

$$\frac{1}{2}\,\frac{\dfrac{W}{C_L\rho/_2S}}{\dfrac{(T-D-\mu R)}{W}} \propto \frac{W/S}{T/W}$$

It will be appreciated from these examples that apart from a few 'inconveniences' such as combat allowances, reserves, trim loads, ground friction, such expressions collapse fairly readily into fairly handleable combinations of the stated ratios but apply to the particular conditions to which they relate e.g. take-off, cruise, combat, with or without stores at landing etc. It is then necessary to relate them to a single datum condition such as take-off so that the conflicts and areas of possible solution are best presented for resolution and agreement. This collation again introduces various ratios of weights and further illustrates the need for statistically based data such as percentage structure weights. Since some elements of an aircraft are fixed (typically payload) absolute weight itself will be involved.

Figure 4 shows the results of such an investigation: range was first presented as a function of T/W, W/S at take-off for several absolute weights: one weight was then selected on cost grounds and other performance requirements added for the selected weight as a series of bounds of thrust and wing loadings. The area remaining unshaded represents a range of possible design solutions which agreed well with point designs derived less systematically.

While the above essentially aerodynamic, and to a degree non-dimensional approach is being applied over a range of weights, an examination will be in progress to secure a single weight breakdown by growth factor methods. Knowing or postulating elements such as the fixed load of the aircraft, the approximate fractional weights of the structure and aircraft systems, an order of fuel capacity, thrust and hence engine weight, it is possible to derive an overall weight which hopefully is not at variance with that emerging parametrically.

3.2 Cost Parameters

Initial engineering estimates of airframe cost are based on various statistical scalings (Fig.5) (combinations of powers of reference weight) allied to costs of avionics, engines, and bought-out items which themselves may be empirical. Such approximations are overtaken by full Estimating procedure as airframe and equipment are defined. The advance activities of production planners and Supply Departments assist this more meticulous synthesis of production and investment cost which is needed to meet formal contract and commercial activities associated with project definition.

As stated, in the early stages of estimation weight does provide some indication of cost via material content and size which governs machining and fabrication manhours but it is far from being as complete an index to the cost of fighter strike aircraft as many seem to believe.

For a given capability many items associated with crew, armament and equipment remain constant in weight and volume requirement as overall size is decreased. Drags, thrusts and many elements of weight are more nearly geared to the square of linear dimensions (the latter increasingly so as minimum gauge considerations appear). Thus, since some overall similarity must apply as size is decreased, volume which is more nearly falling as a cube law becomes increasingly in demand by the intrusion of the square law weights now expressed as volumes, and by the fixed requirements of crew etc.

Assuming that practical considerations of access can be resolved (since areas have decreased) density will be higher and increasing ingenuity and refinement in design will be necessary. Assembly time will not drop as predicted and the greater recourse to more refined structure to beat minimum gauges will be more costly than expected. Accommodation of fuel becomes increasingly difficult leading to more complex tank shapes, greater sealing difficulty and production complexity.

The absolute dimensions available for control surfaces, hinges and actuators become an increasing challenge. Engines may themselves be less attractive at very small sizes because of scale effects on blades and because of mechanical complexity akin to minimum gauges.

The effect of all such factors is to introduce increasing complexity as size is unduly reduced, such that the variation of cost eventually reverses with respect to weight. (Fig.6).

It seems well understood that weight saving exercises introduced after primary definition are expensive but the effect of *excessive* ambition on weight saving in early phases does not appear to be similarly appreciated. It may be that growth factors derived on much larger aircraft subject to such extreme emphasis on performance as to place them on the 1/1 - 'payload' ratio line are responsible.

Part of the project effort in the feasibility and definition phases will be to convince their own Board and the several levels of officials involved that cost estimates are accurate and comprehensive, not only on unit production price but also on research and development. Production price is more readily susceptible to specific illustration through the application of statistics (both gross and detailed) which have been compiled over many years (e.g. by ratefixing) and which remain valid for light alloy aircraft. Research and development costs have been notoriously difficult to assess; with benefit of hindsight initial underestimation is more readily identifiable than any particular inefficiency of execution. Empirical examinations were available giving the variation of R. & D. cost with such factors as design speed, weight, wing loading. Tooling received similar treatment but the scatter was too great for commercial purposes. Data has now been accumulated from later detail recording to enable total estimates to be built up from many elements of design, flight and ground test, tooling etc. which provide the basis for budgetary control on the new project. Such data is very necessary to condition undue optimism that the passage of time and the application of accumulated experience can completely eliminate unknowns and enable a minimum estimate of cost and timescale to be implemented. This could be approached with restraints such as selection of existing equipment and modest initial performance but the usual level of operational demands and the need for minimum unit cost have the effect of continually pressing the 'state of the art' so that development content includes much that is not routine: allowances have to be made for what is achievable in practice so that reference to past records is essential.

4. LAYOUT CONSIDERATIONS

It is difficult to give a general description of activities in the very early stages of design since the process is iterative with a rate of convergence which depends on the severity of the requirement in relation to the experience of the team involved (Section 4 may be found helpful but appeals to hindsight of solutions which have emerged). Apart from the 'scientific' approach a few experienced designers will set an intuitive lead i.e. guess solutions which are appraised, trimmed where too much capability is found, modified or renegotiated where a shortfall exists. Several alternative layouts will doubtless be considered: in such cases it is essential to ensure that technical optimism and assumptions are consistent e.g. that an assessment of (say) wing planform is not confused by irreleventiy different assumptions on afterbody design, or density of equipment packaging. Good communication within the design team is very necessary but possible because of the small team involved.

It is a matter of judgement as to what extent possibly quite legitimately involved differences should be admitted into a study: if too much variation is allowed, the degree of uncertainty is increased and may obscure

the more direct differences. In the final event one particular wing planform may affect fuselage density by (say) matching up better with a particular engine installation or tail arrangement but if in fact the engine installation was changed completely for maintainance reasons and the tailplane was changed for flutter, their original inclusion would not have been relevant. Thus when a variation is being studied it is generally preferable to constrain changes to those which are directly and irrevocably associated with the change, otherwise the whole process could become a technical stalemate.

When a combination of guesstimate, past experience, engineering analysis and cost estimation have specified a combination of weight, thrust and wing area which appear capable of satisfying requirements at acceptable cost, it is necessary to establish physical characteristics in more detail.

Weapon system philosophy has so emphasised mutual dependence of features and functions within an aircraft almost as if to defy any start ever being made, but in keeping with everyday experience a start can be made by defining those parts which are, or appear to be, the easiest and most nearly self-contained.

Where a single satisfactory solution is apparent it should be adopted as a basis: a novel proposal should not be admitted until it is apparent that accepted practice is not adequate. Where several solutions are possible even an arbitrary choice is better than no choice at all: if such a selection is confirmed this is clearly satisfactory but even if not confirmed it is still progress. The essentially iterative process of design, particularly in the early stages of project activity is reflected most visibly between the phases of feasibility, definition, development and production when formal re-assessments of proposals and progress are made with possible re-orientation of requirement, but analogous scrutiny applies down to detail. Filtering of information and adequate communication on the lines of the project/line management matrix described in Section 2 are necessary to ensure this.

During initial definition, project management has to judge when recommendations from one area of activity should be allowed to change a datum established from the balance of many considerations: if a change is made revision of otherwise stabilised areas may occur, causing them to develop their own improvements or deficiencies, prompting further changes or a return to the earlier standard. To avoid such oscillations and since time and manpower are finite, a change may have to be tentatively admitted for assessment in parallel with the main datum. This is a controlled form of indecision which must not be overdone. However, despite the best of intentions, several permutations of 'quasi-datums' will exist in contrast to the rigorous procedures for consistency which have to apply in the later phases of aircraft design and production and which occur naturally in industries having little innovation: aircraft project staff must be able to work constructively knowing that there are several current variations on a datum or with several 'rival' datums, without worrying that work is out of control. Designers seconded from line departments to work with project staff may initially find it difficult not to do this, but once 'acclimatized' help to achieve a balance. They also ease later phases of definition by ensuring continuity of information when the design passes to main departments.

Good suggestions must always be sought and encouraged but it is sometimes necessary to curtail the search for optimum solutions as prompted by individual specialisms: frequently the difference in direct merit - typically weight - between alternatives is found to be small after much comparative work has been done and the overall effect within the accuracy of estimation. It is certainly necessary, but not always easy, to avoid the development of alternatives without much to choose between them in order to try to force out some significant difference between them. This prolongs indecision and denies manpower to more worthy subjects of which the item under discussion is no exception: it can occur after protracted comparative investigation to determine a preference, that overdue assessment in real depth discloses shortcomings which were common to either proposal. For example, even a worthwhile investigation of tailplane frame depth and actuator installation, trading structure weight for afterbody cleanliness, could be detrimental if pursued too far or too long to the detriment of stiffness or its earliest consideration.

Unlike those for civil aircraft, criteria for worthwhile weight saving on military aircraft are difficult to quantify in absolute terms and seem to vary throughout the life of the project. It is saddening that many decisions taken with quantification really seem to be as arbitrary as without it. This occurs where weight or performance gains have to be assessed against a difference in final maintenance, or development slippage etc.

Procedures for more precise quantification under the stimulus of value engineering can be useful but the amount of data required for reaching a decision (for example on wing planks) is sometimes akin to that needed for final manufacture so that again the balance of manpower allocation needs watching.

One area where quantification is helping considerably is in safety and reliability assessment: possible but remote modes of failure, or combinations of failures such as previously led to the embodiment of design features increasing weight and cost without in the event significant effect (or even a detriment) are now unlikely to be adopted.

Having made these introductory remarks and fully realising that in practice a large measure of interaction does take place, various sections of the aircraft are now briefly examined to see how readily they can be made to respond to early detailing while overall work is proceeding. This aspect is important on shared projects where final detail responsibility may be vested in different organisations who will need to satisfy themselves as soon as possible as to the practicality of the weight, cost and timescale promises for their final areas of activity. Action to cover interfaces is important but is a problem readily appreciated by all concerned and thereby more capable of being covered than some within individual responsibilities which are less immediately apparent.

4.1 Forward Fuselage

In its most straightforward aspect the layout of the forward part of the fuselage provides the easiest area for rapid definition in detail: cockpit layout is well defined dimensionally in standard handbooks of requirements and given the number of crew, the type of ejection seat, any particular view or armament requirements, radar and black box sizes, the cockpit and nose fuselage may quickly be laid out and passed into the hands of specialist and detail designers; it has been known to remain at Issue 1 even at first flight. This presupposes, however, either that the inputs mentioned are quickly available or evident, or alternatively that a more complex programme has been correctly sequenced so that anticipatory weapon system studies have been carried out: apart from the possible consideration of ejection capsules, avionic considerations predominate this section of the aircraft which has to embody the results of the most comprehensive tasks of integration in the aircraft weapon system. Most of the issues will be the balance between overall effectiveness against cost followed by trade-offs within and between particular weapon sub-systems.

In the former, typically some aspect of specific improvement in navigation, or delivery accuracy, in optical or electronic view often coupled with weapon size will have to be balanced against the extra cost of providing these, both directly and on the aircraft with growth effects.

Through his team the project designer must draw upon the advice of avionic and operational specialists who, while aware of the merits of the items of innovation or the degree of sophistication they propose, do not always appreciate the associated development risks, the direct and indirect size and cost increases of the aircraft and its possible deterioration because of fixed budget and the effect on other modes of operation.

Such considerations immediately introduce a need for knowledge of the whole aircraft and its unit and development costs so that the effect of the items under consideration can be assessed. A postulation of the environment in which the aircraft is to be used both in peace and war is required but the latter is not always subject to precise definition. Emphasis on flexibility of the aircraft and its equipment is therefore of great importance.

It should not be thought, therefore, that the need for action all lies on the avionic and operational sides; a basic aircraft concept is quickly needed and the quantitative aerodynamic performance effects of extra protuberances can be difficult to estimate.

4.2 Rear Fuselage

The rear fuselage usually houses the engines and is therefore less easy than the nose to consider separately from the rest of the aircraft because of continuity of ducts and intake. Loads and stiffness requirements are high and structural considerations need to be kept in mind from the start.

Like avionics the propulsion system is a most important area in which overall iteration supported by associated specialist skills must be related to fundamental requirements, appealing to the greatest extent to the benefits which variation in engine size and thermodynamic cycle can give when a new engine is permissible. Through weight and balance the engine and its fuel affect more weight in the aircraft than any other source. A new engine will only be admissible if its advantages are proven in relation to the existing engines. This itself will demand studies embracing total costing including R. & D., Production, and Service phases: it may be necessary to assess the merits of single and twin engined versions.

Where absolute cost dictates the use of an existing engine the best selection and proposals for adaptation will have to be made. In any case the principles of installation down to comparatively detail features can materially affect performance through base drag, nozzle and intake efficiency. Anticipatory activities by theoretical, wind tunnel and test rig activities, all related to available flight experience are therefore at a premium.

Choice of engine size is the most important task. Before reliable reheat was available, the selection of an engine of large thrust to cover maximum speed and longitudinal accelerations would inevitably prejudice range and endurance by virtue of uneconomic running when throttled back. Availability of reheat has relieved this situation and is generally used on modern fighter-strike aircraft; providing a welcome flexibility for the aircraft designer even with the high fuel consumption when lit and the larger, heavier jet pipes, burners and nozzles which have to be accommodated at all times. Reheat gives high thrusts for take-off but the smaller engine gives overall economy for cruise and loiter at dry ratings. Supersonic speeds are readily attainable but introduce optimisation procedures for engine sizing, selection of intake and afterbody/nozzle combinations as affected by operational requirements. Pitot intakes have the merit of simplicity and good efficiency for fighters typically up to $M = 1.8$ and are decreasingly adequate beyond, but if allied to short take-off and extended subsonic cruise they need doors and/or variable inlet geometry to aid inlet matching. The inlet may then be associated with geometries giving two shocks for better supersonic recovery around and above $M = 2$. Devices such as convergent-divergent nozzles are claimed as attractive for Mach 2 thrust-minus-drag but are heavy and do not give the best economy in subsonic cruise because of base area and overexpansion. Thus aircraft such as TSR.2 with $M = 2$ capability nevertheless had an aerodynamically simple rear fairing because of physical aircraft balance and an operational radius which involved both subsonic and intermediate supersonic speeds. The Lightning even with initial supersonic emphasis retains a simple rear fuselage shape and a fixed intake: efficient integration with air-to-air weapons and avionic system give required operational characteristics without a complex propulsion system.

In terms of internal thermodynamic parameters, maximum turbine inlet temperatures have always been sought since apart from amortised research and incremental direct production cost they represent maximum energy extraction from inlet air and kerosene fuels with benefit to thrust, radius and overall aircraft sizing.

Increase of compression ratio is another generally desirable aim moderated by feasibility, cost and complexity, engine weight, and on applications with definitive cases at high Mach number (where compression is increasingly provided by the intake once high speeds have been attained). Various proposals for mixed propulsion units have generally not been acceptable on strike aircraft because of logistic objections.

With the introduction of by-pass engines, a further degree of freedom becomes available to the designer, but one again demanding further activity to decide the best characteristics aero-thermodynamically. If by-pass ratio is set high, problems with reheat may occur, at a level decided by the particular form of reheat burner; if sizing is set for take-off, time to height or S.E.P. with reheat, the dry engine may be inadequate in thrust for subsonic dash. If too close an approach to a pure jet is proposed, s.f.c. increases and inadequate radius results. An example is shown with air mass flow as abscissa deciding time to height, and by-pass ratio as ordinate decided by subsonic dash and radius. Realistic body shaping closes the area of possible solution. (Fig.7).

By-pass engines have by virtue of their cooler outer annulus given valuable relief to airframe heating considerations but the cooling arrangements for the engine and its zoning need early definition. Experience gained on the reheat of hotter straight jets is directly relevant to the hot zones of by-pass engines. Greater tendency to intake sensitivity, less flexibility for air tapping and engine re-lighting need attention.

The rear fuselage houses the most important tailplane (usually taileron) control actuators which have to be sized for adequate operating force, rate and stiffness, with good back-up structure and adequate volume for stiff levers, input stages and access. Servo engineers have therefore to contribute to detail design at an early stage.

Fuselage lines and spigot geometry have to be compatible with a wide angular taileron operating range without gaps, and to give a pivot axis/aerodynamic centre relationship compatible with load capability of the jack and with aeroelastic requirements, both static and dynamic. A close integration must exist between this installation and afterbody lines on the outside of the fuselage, and with the jet pipe/nozzle on the inside. Clearances between the engine and surrounding structure, in which the tailplane attachment frames are most fundamental, presents a matter for the earliest discussion between airframe and engine firms, with advice from maintenance engineers, the earliest use of mock-ups and tentative clearance with customer representatives, such as C.S.D.E. in U.K.

4.3 Centre Fuselage

The centre fuselage being by definition adjacent to the centre of gravity necessarily contains fuel which must basically balance about the c.g. and the armament and main undercarriage attachments which cannot be far removed from it. It must provide efficient structural connection between the front and rear fuselage sections, transfer symmetrical loads between wing panels, and pick-up antisymmetrical wing forces for eventual reaction at the tail surfaces. Adequate volume must be provided outside the fuel tanks for control and cable runs and for engine intake ducts on which it will doubtless be considered prudent to allow an area margin for engine development as later changes would be so fundamental as to be best avoided. In view of these many demands for volume, the maximum cross section of the aircraft lies in this section and this, coupled with overall area distribution will be the subject of close attention for minimisation and shaping consistent with realistic clearances and access. Detail deliberations involving operational usage and an understanding of the enemy fire to be encountered are needed to decide the type of tankage and its protection. With operational radius these considerations condition the extent to which fuel tanks are allowed to infiltrate into structural voids not ideal for them. Rectangular section tanks have good characteristics such as low area to volume ratio for minimum vulnerability and minimum bag tank weight. They give good access and have low edge length to volume to minimise sealing difficulties where integral tanks are employed, and are easier for production. Departure in favour of more difficult shapes should be taken only when essential.

The whole of the centre fuselage makes great call on the judgement of the designers in calling upon the specialisms of vulnerability, reliability, as conditioned by maintainability: installation of controls and equipment demands particularly careful technical balance.

It is becoming increasingly common for the undercarriage to be installed in the centre fuselage. In order to help move the c.g. forward after take-off where an aft limit is advantageous for nose lifting, there is an attraction in retracting at least the main undercarriage forward. This usually gives a greater area of secondary structure and doors as opposed to rearwards retraction where some integration with engine access is sometimes possible.

The selection of tyre size, pressure and wheel arrangement will doubtless be the subject of specialist study balancing the advantages of low pressures against disadvantages due to the stowage volume required to accommodate an undercarriage of soft field capability which affects maximum cross sectional area and thereby drag and performance. Checks at the true soft field level, at that for drained grass and tracking, and concrete are appropriate and clearly should be related to the proposed landing and take-off runs. A whole specialist activity has grown up to advise on the number of operations from given surfaces as a function of reaction and tyre pressures.

By exercising geometric and mechanical ingenuity undercarriage bay size and its shape may be tailored to match wheel, intake and tank contours, but the local increases in weight, complexity and development risk must be assessed against overall economy.

4.4 Wing and Tail Surfaces

Wing surfaces are somewhat analogous to the nose fuselage as far as this paper is concerned in that design can very rapidly be supposed to proceed when parameters such as size, planform, thicknesses, control surfaces and high lift devices are defined, but this supposition rests on a great deal of work in the aerodynamic, structural, and aeroelastic areas of activity, reinforced at the earliest opportunity by wind tunnel tests. Since many development tasks and problems of high speed aircraft are associated with low speed conditions, the rapid production and testing of low speed models, basically of wood and thereby capable of ready modification is a welcome capability, now aided by the rapid mechanical production of aircraft lines from master data derived manually. This facility aids the production of metal models for high speed testing.

Due to the thinness of the wings and their frequent location relatively high on the fuselage, it is now uncommon for the main undercarriage to be installed in the wings: this has the advantage that an uninterrupted wing torsion box can be developed with good characteristics for integral tankage.

Some aspects of selecting wing size are covered in section 3.1, but the full treatment of selection of sweep, aspect ratio, taper ratio, thickness, is beyond the scope of this general paper. The issue is one very much dominated, as far as the author is concerned, by the availability of variable sweep which helps resolve many previous compromises.

In an aircraft having a fixed wing it is necessary to decide on a wing sweepback which is high enough with the chosen thickness chord ratio to permit high speed cases to be attained while still permitting an aspect ratio (and where possible a fuel contribution) to allow the radius of subsonic cruise cases to be met.

Absolute wing thickness must be adequate to allow high lift devices to be mounted and operated and, in terms of chord, for the chosen aerodynamic flow patterns to be made manifest for take-off, landing and manoeuvrability. The choice of aspect ratio and thickness introduces absolute weight contributions which have to be fed into overall performance assessments. Selection of taper ratio is a balance between avoidance of tip stall and installational limits which occur with too great a taper, and excessive weight if insufficient taper is adopted.

The extent and complexity of high lift devices is dictated by landing and take-off requirements plus manoeuvring within the speed range where the stressing weight and aerodynamic effects such as trim are acceptable or can be made so. The importance of view on the approach will condition the choice of single or double slotted flaps.

Section characteristics are a specialist area embracing theory and testing in which it is usual for a 2-dimensional basis to be first understood and these related to possible planforms with various combinations of camber chordwise and spanwise for which computer programs now exist. In cases where sweep and thickness justify refinement, techniques initially developed for airliners are relevant to securing good root flow characteristics.

The wing will doubtless be called upon to carry overload stores demanding strong points and wiring; any fuel tankage will need pipes, pumps or air supply and gauging but otherwise it should be possible to keep the wing simple with good structural efficiency, running flap and slat drives within the secondary shroud structures.

Modern structural computing techniques are particularly useful and worthwhile optimisation of the main box can be undertaken. Production considerations on the choice of the number of spars, and skin reinforcement as affected by sealing can be introduced.

A typical selection is between fabricated skin-stringer wing skins which to be competitive on weight with integrally milled skin panels needed to be in lithium based light alloy: integral construction was selected after preliminary forming tests of the latter material showed cross grain difficulties in thin gauges which would have prejudiced the stringer roots had a fabricated wing panel been adopted.

In all such considerations it is obviously essential that the project engineer consider the material in a form in which it is usable, and that the sizes of sheets assumed (for example) are compatible with what will be supplied: a material which appears attractive on a general specific strength or stiffness basis may not be so attractive in thin gauges, and if sheet sizes are restricted, the provision of extra joints will offset basic advantages.

The availability of Numerically Controlled Machining has made various refinements which would not have been admissible ten years ago both possible, and economic: Titanium flap tracks are one example.

In wing design (and generally) it is still the procedure to derive scantlings on the basis of strength requirements, obtaining influence coefficients as a most valuable by-product of digital calculations. By allying these with mass distributions, including external stores and various fuel states and with aerodynamic lifting surface derivatives, wing flutter speeds, and static aeroelastic effects can be checked, any adjustment to skin thickness made and realistic weights derived. Static aeroelasticity and changes in flow pattern at

higher incidences give valuable relief to wing bending loads and also ameliorate the basic swept wing aerodynamic shift with Mach number. If reliably known these effects can materially assist basic layout: a series of Wind Tunnel tests in the Project Definition Phase can check the characteristics of deformed wing over a range of normal coefficients.

Although much of the foregoing may appear to be associated with detail design phases, an adequate anticipation of satisfactory aeroelastic characteristics of the wing and tail surfaces in the earliest phases is important and possible with modern methods. Similar considerations apply to the tail surfaces: tailplane spigot orientation is a fundamental parameter in rear fuselage layout; fins and tailerons have to be quite large in comparison with wing area and their being at the rear of the aircraft demands that their weights need to be realistic for balance also.

The wing to tail relationship, and their settings on the fuselage are some aspects of overall integration covered in the examples of Section 4.

Wing body angular setting is typically a balance between high lift, lateral handling and vision on the approach on one hand, and high speed pitching moments on the other. Anhedral and sweep are coupled by lateral stability considerations. Simulator checks are desirable.

Longitudinal wing to fuselage spacing may either be pressed towards a minimum to reduce fuselage weight but needing a large tailplane (possibly subject to non-linearities of downwash and to store and flap buffeting) or for similar longitudinal stability margins, a smaller tailplane on a longer and therefore heavier fuselage may be selected. In the latter case ground clearance angles may dictate a longer undercarriage which will be heavier unless extra complexity is admitted but operational factors such as the installation of armament may be decisive for undercarriage length particularly on a small aircraft.

It is general, but not universal practice, to attain as large a vertical spacing as possible between the planes of wing and tailplane. A wing mounted high on the fuselage helps directly in this respect, but with high sweepback extra anhedral is necessary in such a case because of body wing interaction and this is not helpful longitudinally. The anhedralled tip of the TSR.2 wing illustrated one manner of resolution with a structural penalty for the skin joint and kink rib.

It may be thought, and correctly, that aircraft design is not a matter of deriving several components and then assembling them in a manner reminiscent of a plastic toy construction kit. However, it is the function of the project engineer to 'coagulate' as many areas as possible so as to minimise the number of permutations of unknowns and it is worthy of note that several successful families of aircraft have appealed to not dissimilar concepts in development and production (Mirage and several civil examples, and, evidently during competitive phases in the Soviet Union:

Fitter, swept wing; Fishpot, tailed delta

Faceplate, swept wing: Fishbed, tailed delta)

Possibly more could be done to exploit this concept to help reduce development cost and timescale, and to reduce production costs.

5. SOME EXAMPLES OF OVERALL SYNTHESIS OF DESIGN

Some illustrations as seen in retrospect of the way in which aircraft have fitted together, may be helpful since it is difficult to describe in general and in an interesting manner a procedure which is iterative and highly responsive to the particular conditions which apply, both in terms of detailed requirement and also to the available state of the art.

The P.1 which preceded the Lightning, was designed when reheat was not fully proven but nevertheless full supersonic performance was required. Twin engines mounted above each other in the rear fuselage gave a frontal area compatible in shape and size with a seated pilot and flat-stowed nose undercarriage: even when allied with a fairly large thin wing of high sweepback, supersonic speed could be attained with dry thrust. The installation of the engines in the fuselage was accommodated by large load carrying doors.

The twin ducts made a mid-wing attractive since its centre section could pass between them. This arrangement was also compatible with adequate wing to tailplane vertical separation and a practical undercarriage retracting into the wing, which also housed the fuel. These measures were both intended to minimise body frontal area on the basis described. The undercarriage had to be long enough to meet the large ground incidence associated with highly swept wings: its stowage within the thin swept wing demanded considerable detail ingenuity.

A high wing would have rendered the undercarriage impractical; a low wing would have given poorer or unacceptable handling characteristics.

The engine ducts were necessarily long and contoured but worked well without development problems. The outside of the fuselage (although once unkindly described as resembling a suitcase) nevertheless gave a simplicity of shape for construction in contrast to the more exotic shapes which quite incorrectly became synonymous with 'area ruling'.

The wing planform chosen was a delta with the rear centre notched out. This section is relatively inefficiently loaded at supersonic speeds and its wetted area is better used for the tailplane which by being all moving (introducing power controls) and by exploiting wing aeroelastic effects could be quite small. Ailerons were positioned across the remaining transverse wing trailing edge, with their rolling force on the effective wing flexural axis giving minimum elastic penalty in roll and no reversal.

Initially nose apex flaps and trailing edge flaps were used but experience showed the latter to be adequate.

In order to reduce development risk, a plain pitot entry at the nose was used (a position often chosen by aircraft design teams extending Mach Number capability).

When operational capability was extended the pitot entry with a radar lip was replaced by a two-shock circular intake with a radar dish installed in a centre body. Individual 'black boxes' were avoided by using the complete centre body as common pressurised surface. (Fig.8).

The opportunity was also taken to simplify nosewheel retraction into a single rotation by using the centre body support as the housing for a vertically stowed nose wheel: it also gave a secondary air inlet for the higher thrust engines available.

In 1956 in the initial studies of a series finally leading to the TSR.2, efforts were made to relieve the weight and installation penalties of fuselage mounted engines: by then reheat was accepted as reliable, giving worthwhile economies overall but imposing detail constraints because of the associated higher jet velocities and temperatures. In an endeavour to revert to an earlier form of conventional solution, under wing nacelles were investigated (Fig.9) with a tailplane raised above the jet efflux: however, even with an exaggerated 'T' tail (which needed bracing in the wind tunnel) undesirable pitch-up still occurred and the layout was dropped. The most unconventional arrangement was then evaluated: this was a tail-first configuration with 'conventional' (i.e. wide-coupled) canard to wing spacing, on which a divergent chase between control requirements for nose lifting and stability followed. (Fig.10). The need for the canard to stall after the wing inhibited the total low speed lift to be secured offsetting to some extent the direct benefit of the upwards trimming force from the canard. Although S.T.O.L. was needed, increases in wing area were not desirable because of gust response at low altitude and high speed and this approach was discarded.

Thus, in the event, a rear fuselage engine installation was again adopted, (Fig.11) at first with underwing inlets which were discarded because of development risk in favour of forward mounted side intakes. The consequent increase in inlet duct weight, extra tail volume to compensate the destabilising effect of intake lift, and overall size increase, had to be accepted.

In 1959, pending the introduction of by-pass jets, engine carcasses, intermediate and final jet pipes and nozzles were all hot, and therefore sufficiently heavy to confirm engines at the rear to at least minimise structural heating and weights: engines set more centrally i.e. nearer the c.g. have still to be associated with a similar weight of intake and inlet ducting since reasonable intake settling length has to be provided, and have additionally longer hot jet pipes to take the jet efflux clear of the structure at the tail end.

The F-4 and Jaguar (Fig.12) have been exceptions to this formula with structural provision on fuselage and tail against reheat pressures and temperatures emanating from nozzles some way forward of the rear fuselage extremity.

The most usual arrangement installation for engine of fighter-strike aircraft is for them to be located as near to the rear of the fuselage as possible and generally for this itself to be as short as possible so that balance can be met without too long a nose. Depending on the pressure of the operational requirement for range and associated supersonic performance, it will be necessary to depart from fuel tanks within the centre fuselage and wing only and to allow these to extend rearwards in the fuselage: because of the heavy engine weight of rear fuselage engine installations the nose fuselage tends to be fairly long with space (comparatively) to spare in the front centre fuselage (typically between the intakes and front ducts). Thus, if fuel can be accommodated in the rear fuselage, a balancing volume can usually be found in the front giving double advantage for fuel to the rear of the engine face, but calling for special attention because of hot engine spots and air system pipes where these have to be adjacent to fuel tanks.

On TSR.2, demands for long range including sustained supersonic components led to a very high volumetric usage being developed with extensive integral fuel tanks in wing and fuselage, the latter extending around the engine installation and thereby demanding special attention to fire detection, containment, and suppression. Early testing was essential.

After much deliberations, c.g. considerations when carrying underwing overload stores led to a forward-retracting main undercarriage on TSR.2 requiring considerable ingenuity to allow it to be stowed adjacent to the intake ducts and armament bay within acceptable cross sectional area, and leading to one of the few development modifications needed in the short but comprehensive flight test programme.

Short take-off requirements led to stringent weight control measures, particularly in the rear fuselage because of dry c.g. considerations.

A variable sweep wing was considered in early studies for TSR.2 but was rejected on grounds of timescale and development risk: in 1958 there was insufficient data on large heavily loaded and comparatively slowly,

intermittently moving bearings. Gap sealing was aggravated by the belief that it was still necessary to translate the wing bodily as its sweep changed. The best fixed wing balance between lift for take-off and landing, a wide range of cruise conditions sub and supersonic, and low lift slope for minimum gust response was found to be a 60^0 delta with full-span trailing edge flaps. This was made possible by extensive wind tunnel and simulator verification of tailerons i.e. tailplane panels giving pitch and roll. The use of 60^0 leading edge sweep and all-moving tailplane panels followed the satisfactory experience with Lightning on separate tail panels moving symmetrically.

This earlier experience has fed into later aircraft such as M.R.C.A.; but in the intervening period the status of variable sweep has itself changed, so that it may now be adopted with confidence.

Tests have been carried out on representative bearings, in the case of B.A.C. first by a simple swinging arm rig subject to cycles of loading over a range of sweep positions which gave good clearance of the bearing surface and its immediate housings followed by a more comprehensive rig (Fig.13) which gave a fuller representation of adjacent structure and a check on stiffness and stress distribution. Both mechanical and pneumatic seals have been investigated, the former involving material development. Extensive tunnel testing has developed slat and flap configurations and confirmed trim and stability, with handling checks by simulator.

The necessity for wing translation with sweep may be avoided by two arrangements: that developed in America has an outboard pivot in the wing and large fixed apex. Pivots adjacent to the fuselage side walls without a fixed apex are a European solution with various merits which need not be pressed in this general paper.

Variable sweep wings must be regarded as any refinement is regarded, namely the source of additional development, weight, cost and complexity as seen by itself but which in the correct application is capable of conferring overall savings which more than offset these local disadvantages: were it otherwise such devices would not be selected. (Variable pitch airscrews, retracting undercarriages are earlier examples).

The compensating and overriding advantages of variable sweep are a high lift mode associated with full flap and slats and forward wing position with high aspect ratio. High aspect ratio for loiter and cruise gives economies in overall size while still permitting rapid acceleration. High speed cases are covered with wings swept with excellent ride characteristics throughout.

6. INTERNALS

Having given some indication as to how the aircraft is derived, a superficial account of internal overall definition may be ventured.

In practice this draws upon the work undertaken in the several individual areas and is in parallel with it.

6.1 Systems

The designer has some areas to a degree defined for him, for example:

Operational factors earmark certain locations for armament, for radar dishes and other aerials: crew arrangements will dictate displays, seats and controls, vision, heating and cooling: engines are associated with their own pumps, and with various off-takes for secondary air systems, air conditioning, gear boxes, starters, constant speed drives, pumps and alternators, and need main ducts of given size running forward to the intakes.

Booster pumps and gauging in fuel tanks are dependent on tank shape and layout; these, as described, are dependent on first iterations of duct lines, with parallel assessment of protection and explosion suppression.

If the aircraft has a variable sweep wing, volume for the wing root, pivot, actuator and seal will need to be provided. Even if an armament bay is not provided, ejection units for stores will be needed with appropriate structural attachments and wiring runs.

Hydraulic jacks associated with undercarriage, airbrakes, and power controls are necessarily compatible with the kinematics of undercarriage retraction and surface rotation as conditioned by afterbody lines and junction fairing. Equipment requirements can be expressed as a series of bay sizes and since overall area plots soon become available it is possible to check that the volume available from performance considerations is adequate i.e. to carry out an overall check on density (Fig.14). Since space is at a premium, the volume allowed for accumulators and reservoirs, jacks and power controls demands an early check by the systems engineers concerned, relating rates of movement, hinge movements, system pressure etc. to pipe and cylinder demands. This leads back to the overall description of the aircraft being required.

Attention has to be given to the volume requirements of aircraft systems not only at the power sources and usage but to the pipes and cables which connect them: to connect the various areas of source and usage, typical longitudinal runs in the nose fuselage are in side consoles, under the cockpit floor and alongside the nose undercarriage. In the centre fuselage, natural runs with good access exist along the lower centre line in a duct or armament bay and at the top as a spine. To achieve a connection between these at the front is relatively simple by runs at the rear of the cabin pressure bulkhead, but similar interconnection at the rear is generally more difficult as may be the lateral branching of systems from the central duct to the engines on a twin engined aircraft.

Overall reliability examinations will be needed to check where sytem duplication is justified. One example is afforded by an examination of single and twin engined aircraft in which two designs were completely costed over a range of fleet sizes and attrition rates, see Figure 15. A parallel examination was needed to establish the most likely attrition rates so that a conclusion could be drawn.

Throughout the derivation of the equipment layout the aim is to achieve the lightest and simplest layout overall including structure and reconciling to the greatest extent the requirements of maintainance and survival which often conflict: the detail attention of later stages has to be anticipated bearing in mind production (for example the balance between long 'clean' runs which may in practice need special breaks and access panels on one hand, and more tortuous installations which make concessions to structural continuity, on the other).

6.2 Structure

The structure of the aircraft has to pick up, collect and redistribute the inertia loads acting on each mass in the aircraft and to ensure they react the aerodynamic and/or ground forces in the most efficient manner with desirable or acceptable distortions.

It must balance out as locally as possible those forces which naturally do so, for example, bursting pressures from the pressure cabin and those elements of aerodynamic surface pressure which have no resultant (e.g. ribs).

Certain major structural units have to exist in order to pick up point reactions and therefore give the structural designer a start in laying out the structure and adding the necessary longerons, spars and ribs:

- seat attachment frames, bulkheads
- nose and main undercarriage frames and shear webs
- taileron spigot, frame and shear webs
- wing attachment frames and bulkheads
- store attachments
- slat, flap, spoiler attachments
- hook, brake parachute, airbrake attachments
- engine attachments

Others are required to contain pressure or fuel, or to give aerodynamic continuity:-

- skins and engine ducts
- wing tank closing spars and ribs
- tank and cabin pressure bulkheads
- engine compartmentation webs and bulkheads

Some exist as edges and corners:-

- canopy, door, bomb cell edge members
- longitudinal (or spar wise) edge members
- surrounds to doors and access panels.

The object in laying out the structure is to exploit items such as those listed to perform as many duties as possible where it is efficient to do so, remembering that it is basically the skins which will try to make the main transference, and applying some caution where the combination of integral tanks with major concentrated loads is contemplated.

The extent to which some items should be exploited needs judgement and an early check by computation: for example, the extent to which even stiff tunnels surrounding engines will pick-up load from the tailplane spigot via the tailplane frame and carry it near the neutral axis, as compared to the outer skins, needs checking.

Conversely falsework adjacent to a longeron will strain with it unless a degree of freedom is introduced.

Some aspects of the important issue of material selection were covered under wing in Section 4.4.

There is considerable advantage in having a clear picture of the structure in order to aid numerical analysis and to ensure adequate load path continuity. The production planners and jig and tool designers will also appreciate a presentation which gives an overall impression prior to receipt of normal working drawings.

7. CONCLUSION

The project definition of the aircraft (Fig.16) is seen to be a complex iterative process in which managers and designers draw upon past experience and the advice of specialists in fields such as avionics, armament, propulsion, aircraft system, structural, production, operational in order to obtain the best balance of performance in all its aspects, cost and timescale.

This paper will, it is hoped, have set in perspective the idealised concepts frequently presented for the integration of a weapon system, wherein outputs from groups such as 'avionics', 'propulsion', 'systems', 'structural' and 'requirements' flow into a central co-ordinating box: particularly in the early stages considerable interaction takes place but it is the task of the project designer to consolidate what he can and render the process convergent on to a consistent datum proposal as rapidly as possible. Different activities mature at different rates and some have therefore to be shielded from potentially short period fluctuations, otherwise no viable output from such sources would be achieved.

It appears that aerodynamic considerations initiate most major features of the aircraft being quite fundamental, and that further ingenuity by mechanical and structural engineers implements these decisions in detail. Various flexibilities are now available, reheat, by-pass engines, variable sweep, basically to assist the designer but introducing more effort to ensure that they are correctly chosen.

At the stage described, the aircraft has merely begun its design process in serving as a datum for detailed verification by specialists, including detailed cost estimation.

ACKNOWLEDGEMENTS

The author wishes to acknowledge British Aircraft Corporation's permission to present this paper and the examples therein: he regrets that pressure on personal time precluded the inclusion of more of these. The views expressed in this paper are those of the author personally and not necessarily those of B.A.C.

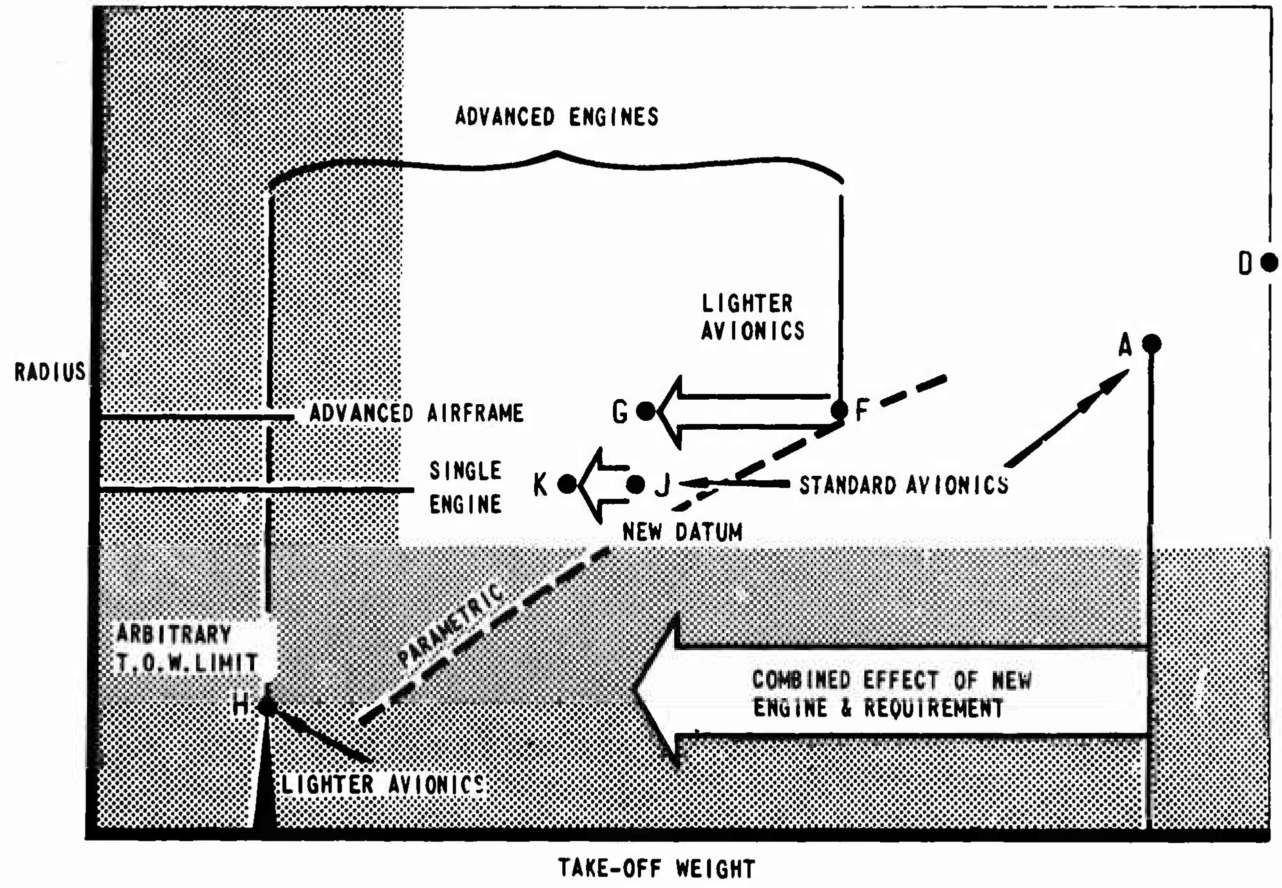

Fig. 1 Parametric study as a guide

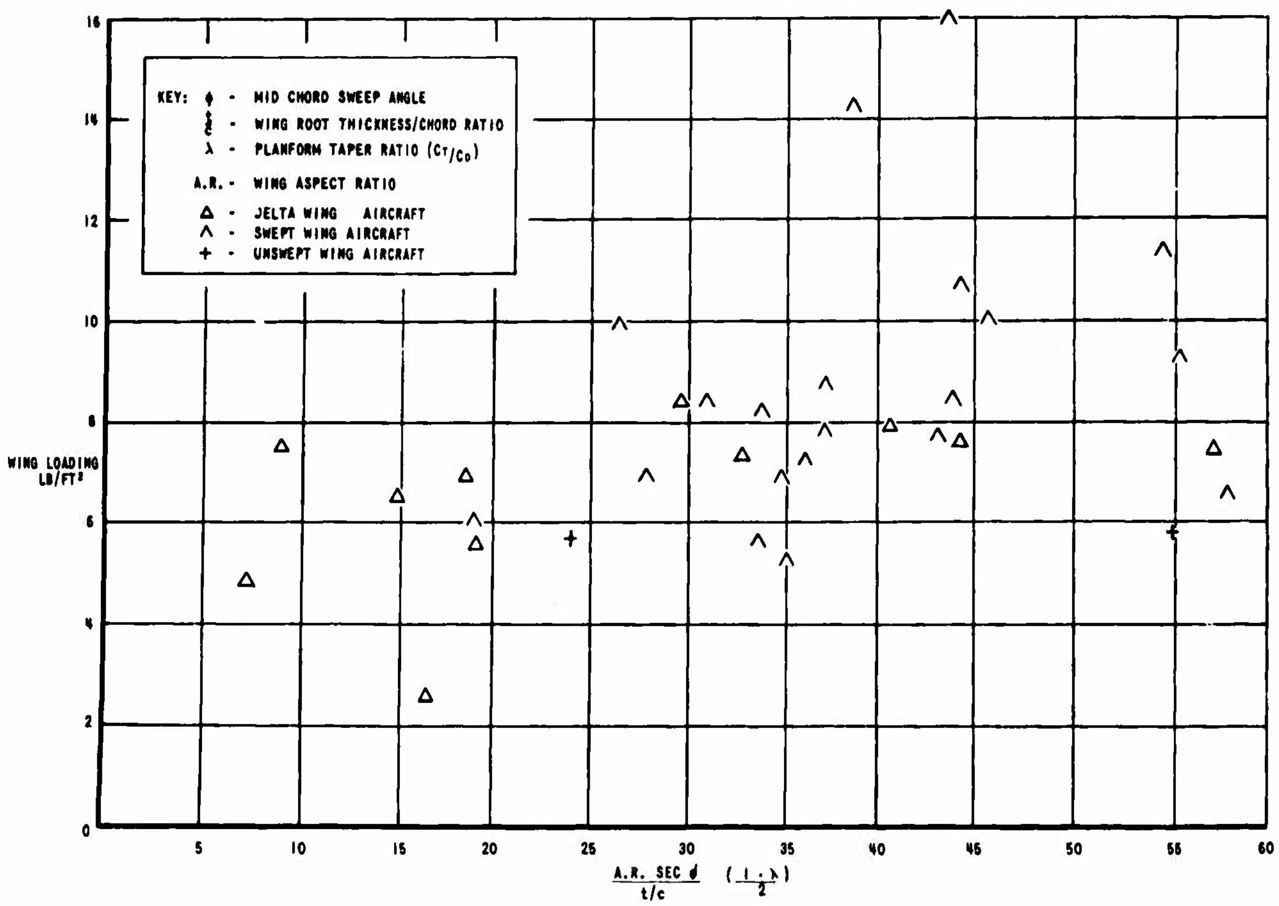

Fig. 2 Wing overhang ratio

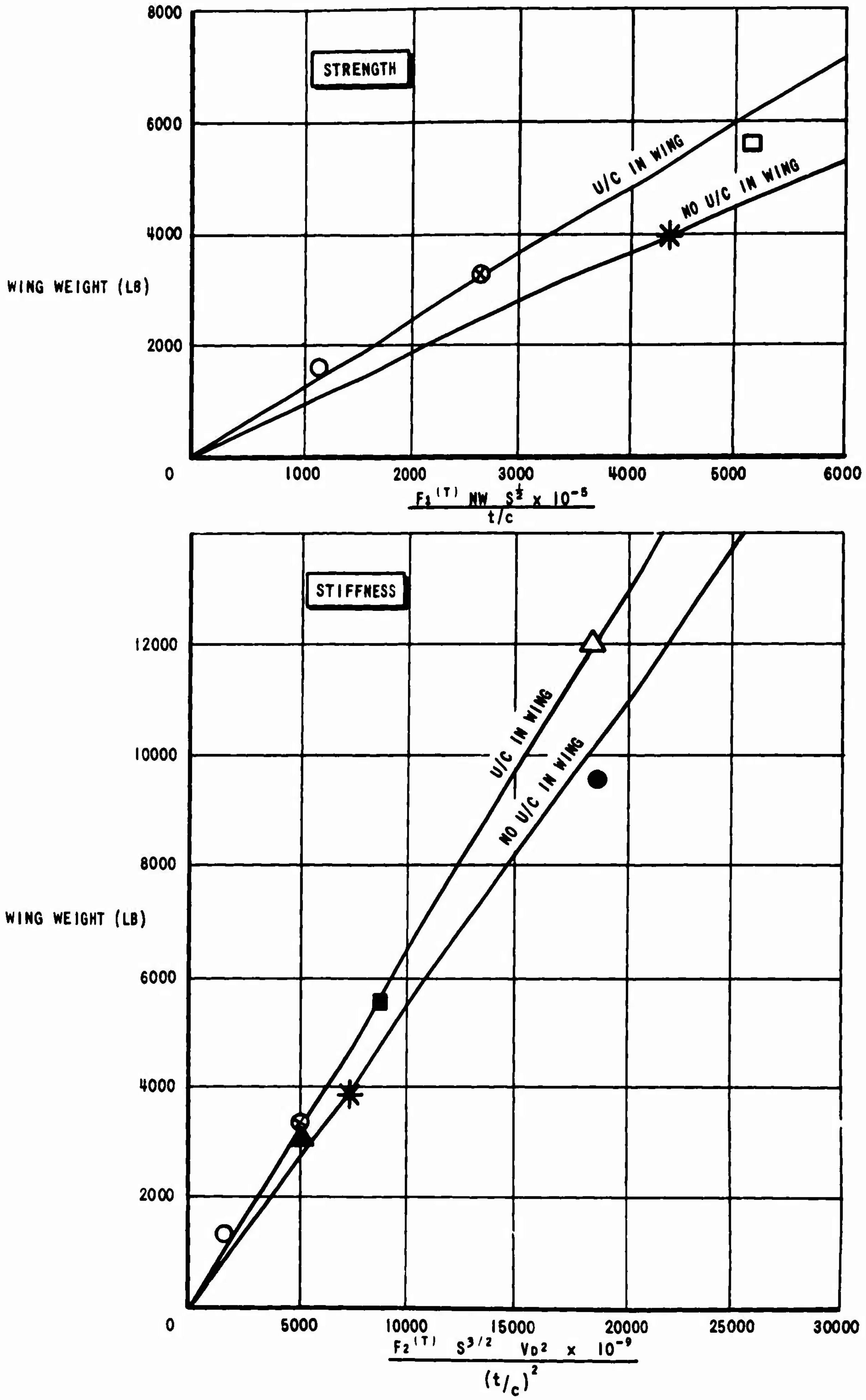

Fig.3 Correlation of delta wings

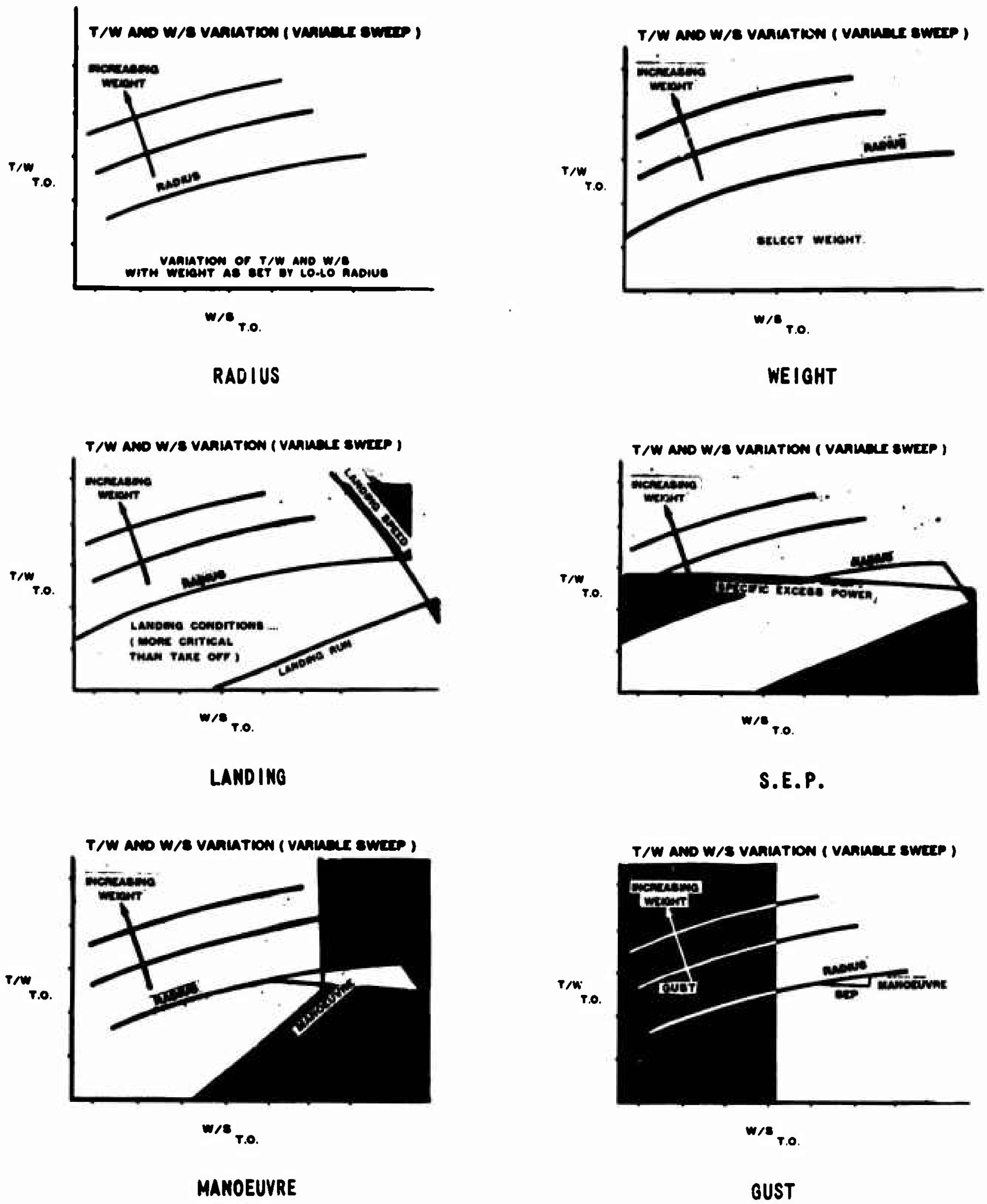

Fig.4 Thrust to weight ratio and wing loading

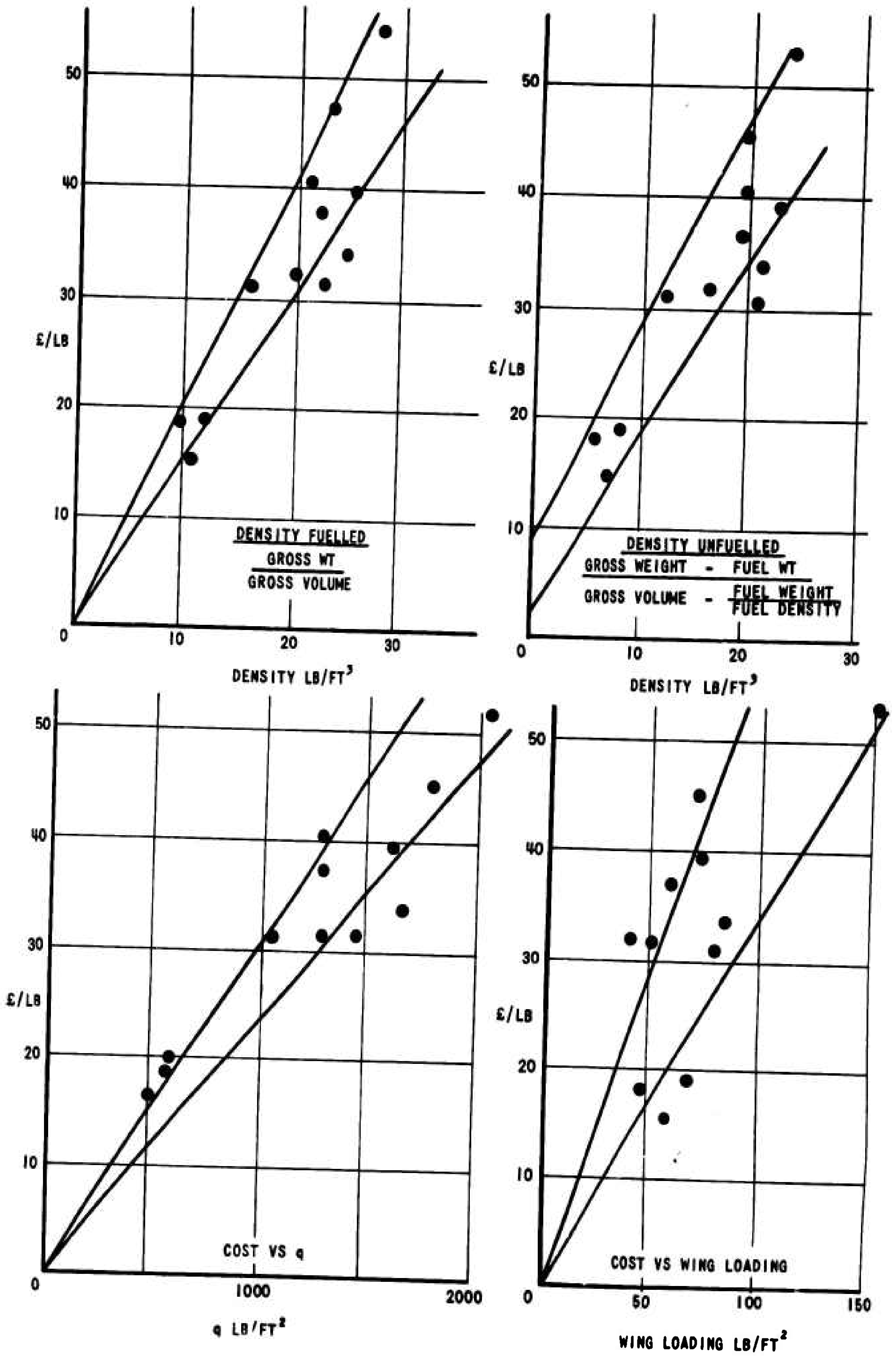

Fig.5 Initial unit cost estimate

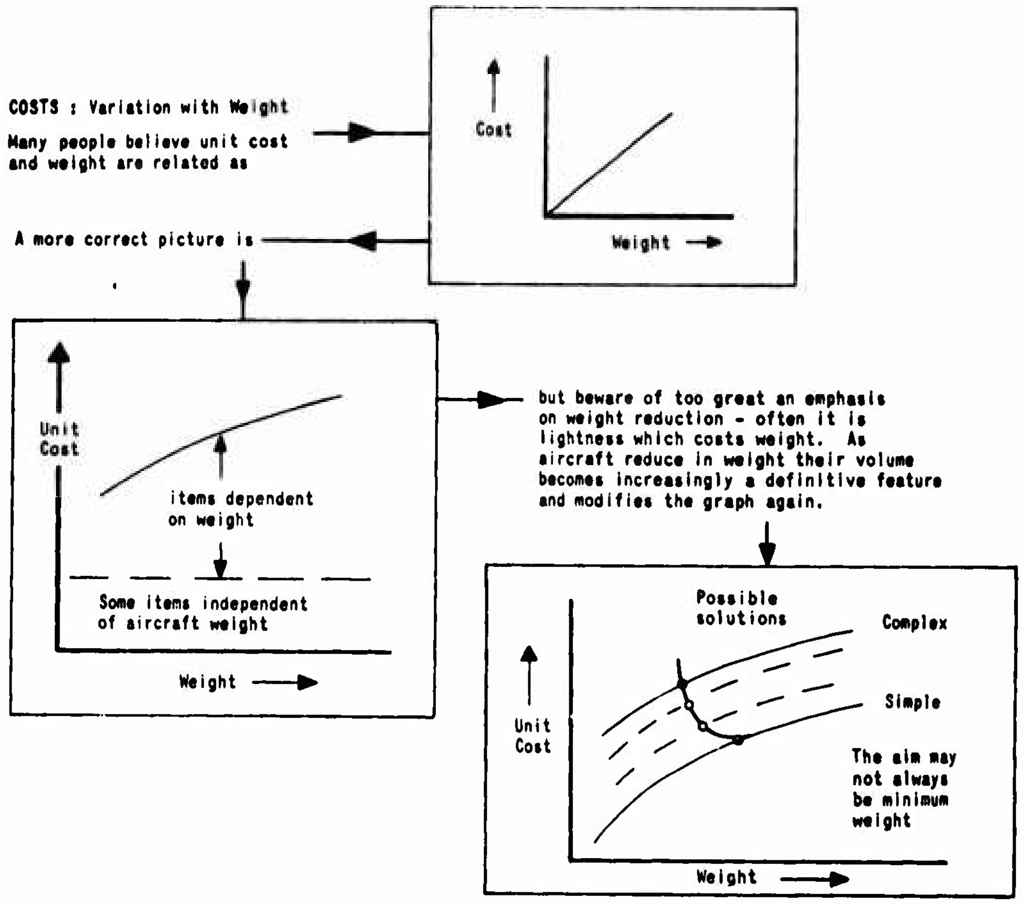

Fig. 6 Costs: variation with weight

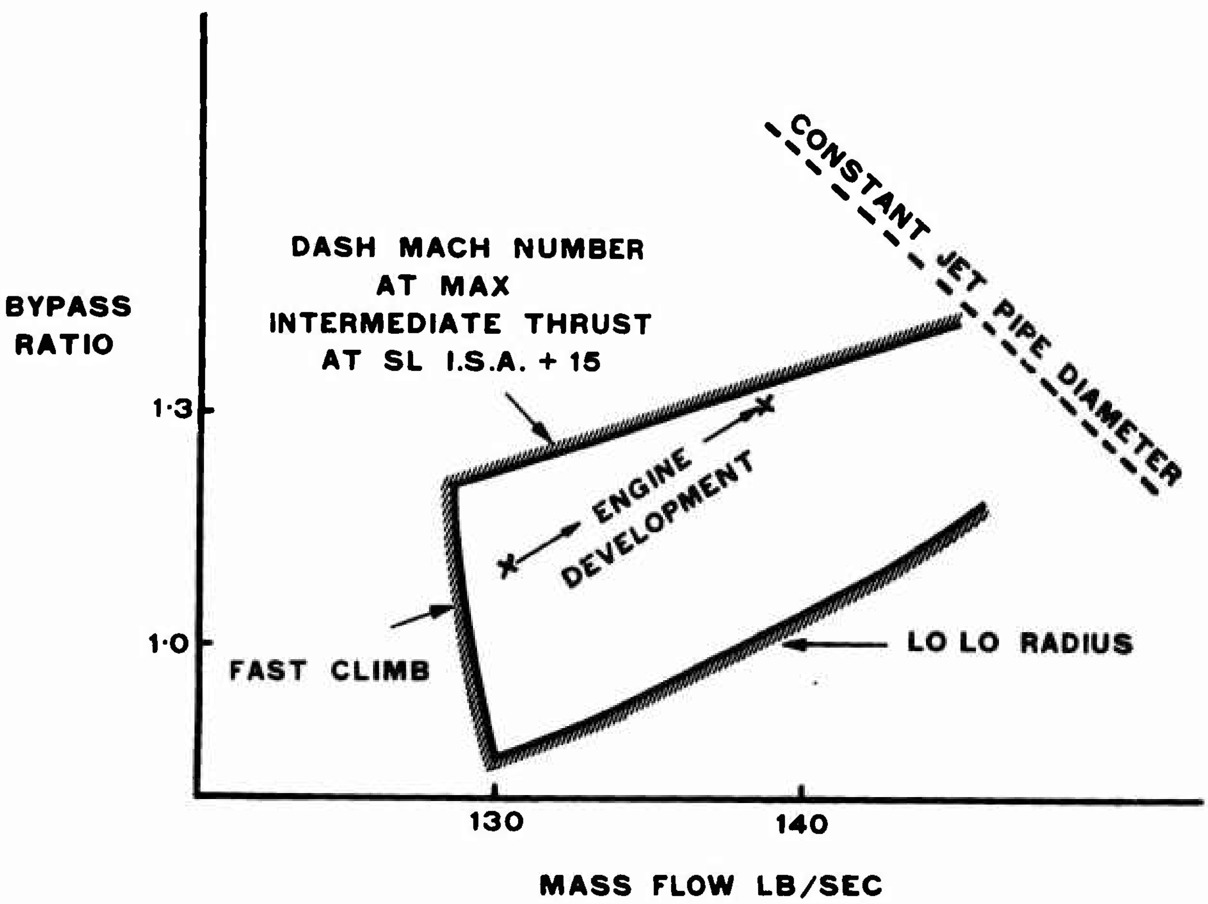

Fig. 7 Bounding values of engine design point parameters

Fig.8 Lightning Aircraft

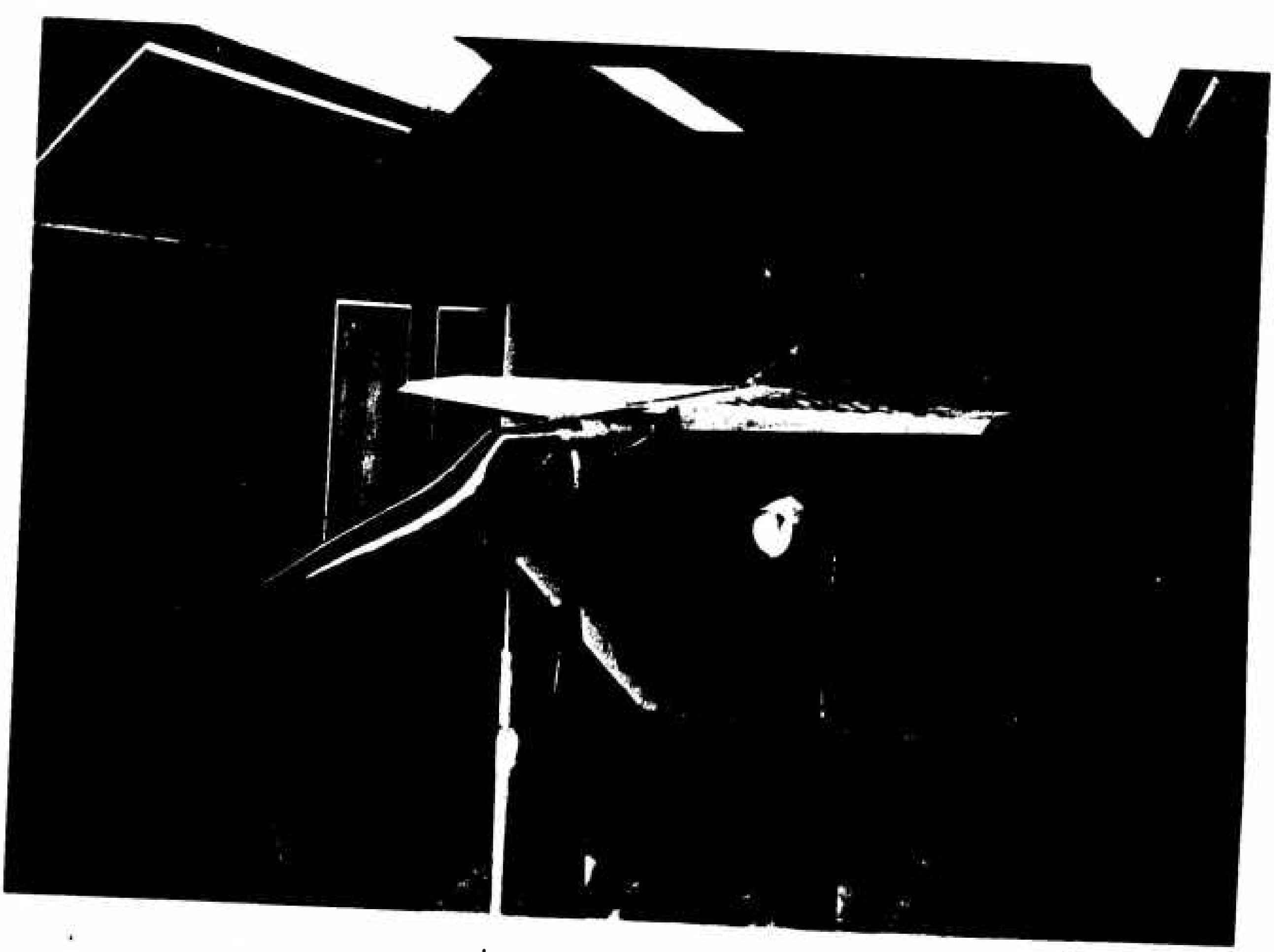

Fig.9 Model with underwing podded engines and high tail plane configuration

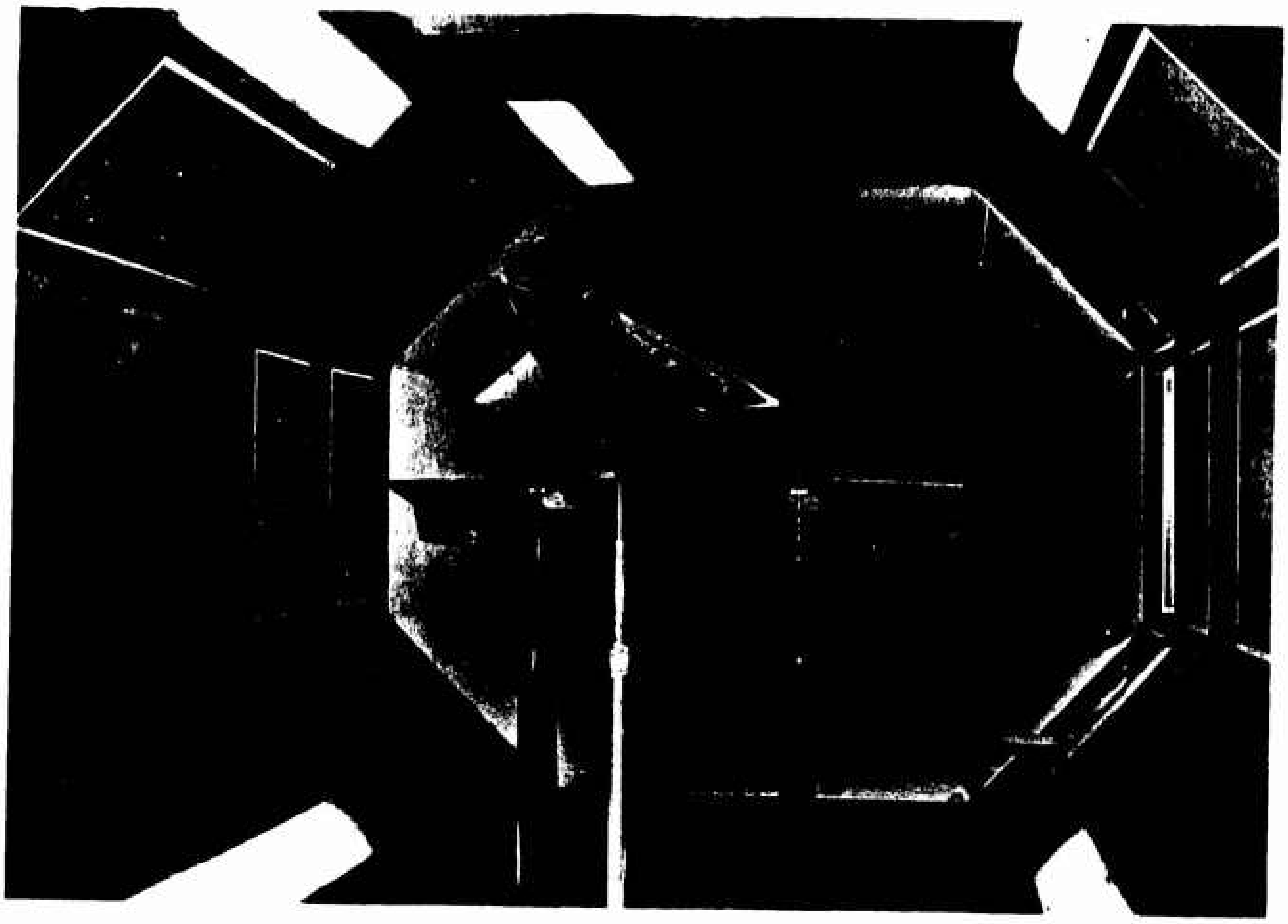

Fig.10 Model with canard layout

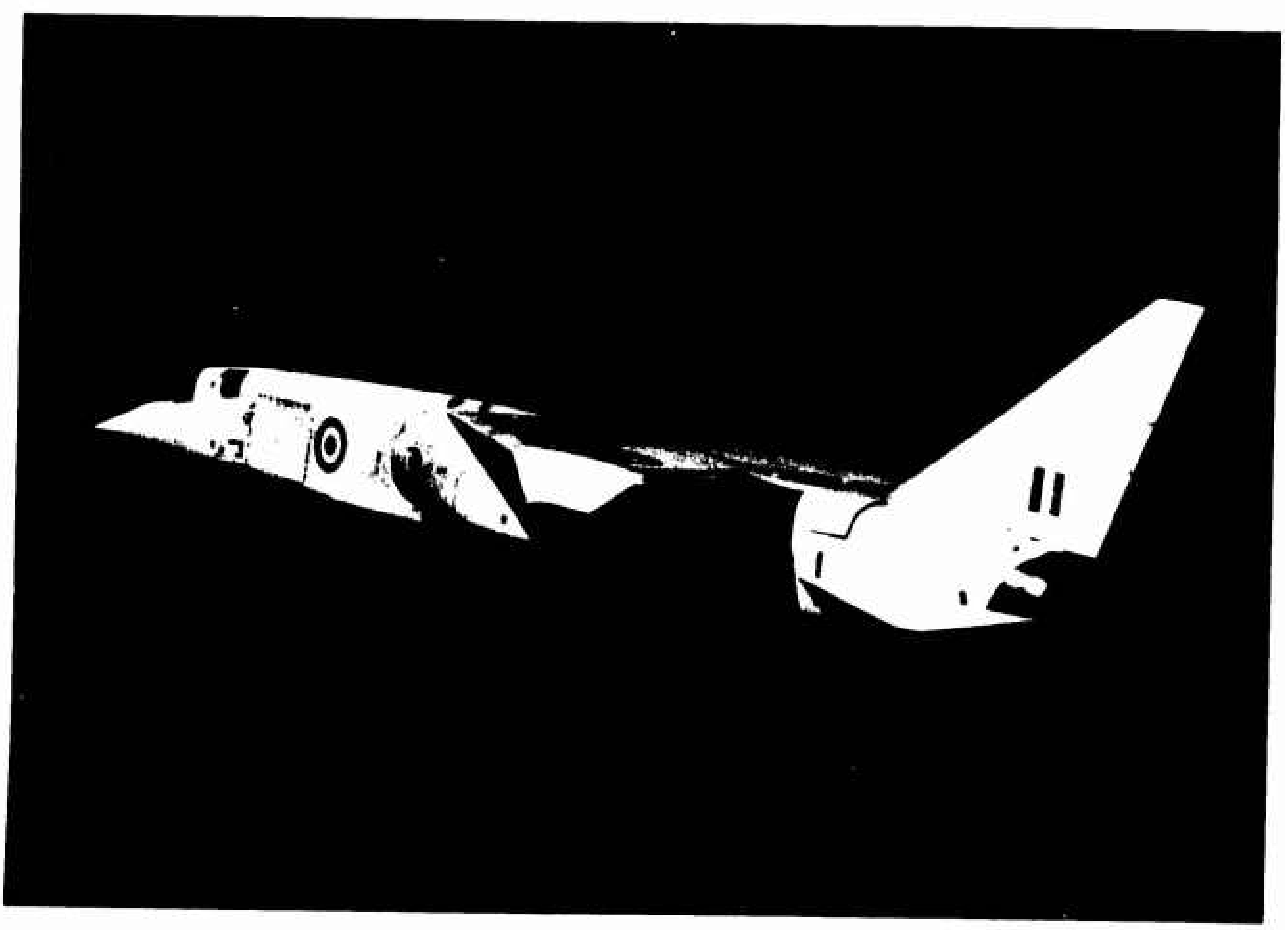

Fig.11 T.S.R.2 Aircraft

Fig.12 Jaguar Aircraft

Fig.13 Structure test rig

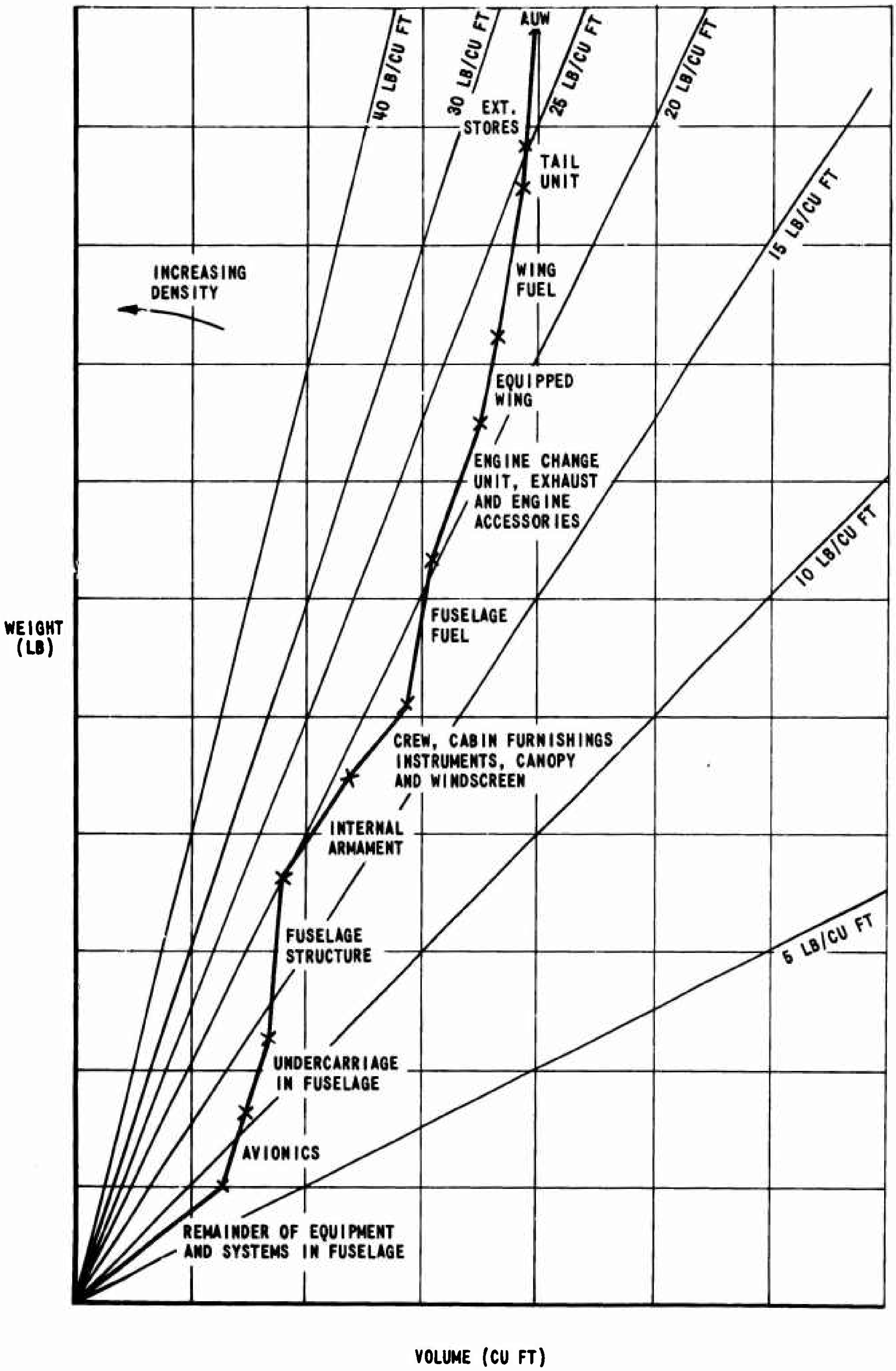

Fig. 14 Aircraft density: typical strike/fighter aircraft

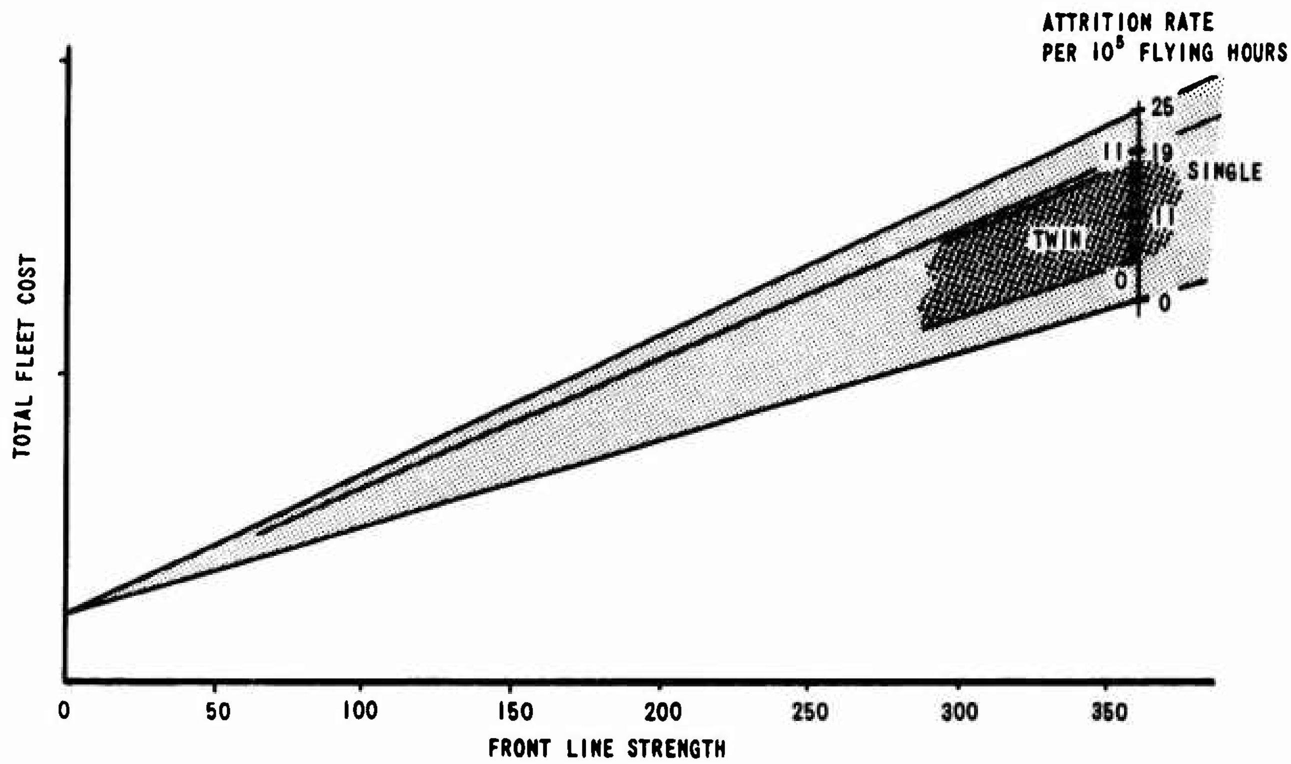

Fig.15 Total fleet cost

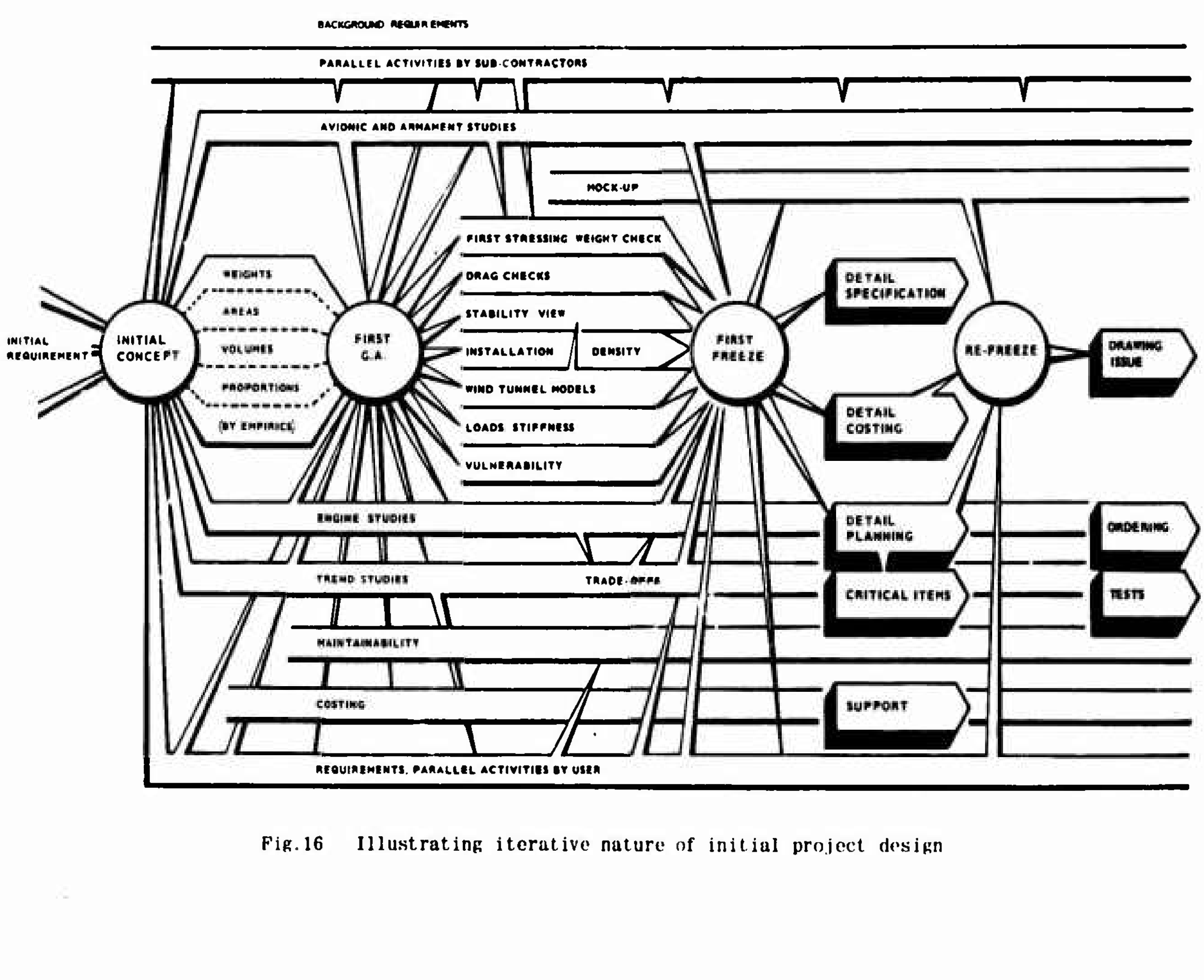

Fig.16 Illustrating iterative nature of initial project design

AVIONS D'ARME

PROGRAMMES ET REALISATIONS

par

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AVIONS D'ARME

PROGRAMMES ET REALISATIONS

H. DEPLANTE

Directeur Général Technique des A.M.D.

Le Général FERGUSON dans un article récent intitulé "Challenge for Survival" a jugé bon d'attirer l'attention des constructeurs d'avions Américains sur la rapidité du développement des matériels aéronautiques Russes et Français (Marcel DASSAULT en particulier). Nous avons été particulièrement flattés d'avoir retenu cette attention et mérité ces éloges.

C'est ainsi que j'ai pensé que vous pourriez être intéressés de recevoir quelques précisions sur le développement de la création du matériel français d'après guerre ne serait-ce que pour comprendre dans quelle ambiance de compétition nous nous sommes débattus.

DASSAULT, Société libre, opposée le plus souvent à des Sociétés Nationalisées, a dû plus que toute autre de "combattre pour survivre" ; de cette situation sortirent les excellents résultats qui confirment pleinement les conclusions du Général FERGUSON.

Le choix des matériels aériens de notre Armée de l'Air (cellules, moteurs, équipements ou armement) s'est fait, en général, non "sur du papier" mais sur des réalisations.

Comme vous le verrez, l'Etat bien souvent imposa la compétition. Toutefois, en de nombreuses occasions, il ne put s'opposer aux propositions logiques faites par un constructeur dynamique, toujours prêt à se battre pour garder la première place, toujours prêt à réaliser, dans un temps record (en P.V. si nécessaire), le matériel ainsi proposé (M. M. Dassault).

Il faut reconnaître toutefois que, depuis un certain temps, à l'occasion des coopérations internationales, on assiste à la tendance fâcheuse du choix "sur dossiers", ce qui risque de réserver des lendemains qui chantent.

I - LES SERVICES OFFICIELS ET LE CONSTRUCTEUR -

La création des matériels aériens français de l'après-guerre s'est faite dans une excellente atmosphère de coopération entre constructeurs et Services Officiels (D.T.I., S.T.Aé, E.M.A.A.). L'atmosphère de suspicion de l'avant-guerre avait pratiquement disparu. Un climat nouveau de coopération était né.

1.1 - Sur le plan des Services Officiels -

On peut porter au crédit des Services les points suivants :

- Souci de l'efficacité -

. Pas de lourds comités neutralisant l'activité du constructeur.

Les Services prirent le soin de désigner des responsables de qualité pour surveiller les réalisations : un ingénieur de marque dans le cadre du S.T.Aé et un officier de marque dans le cadre de l'E.M.A.A. Après une ou deux expériences convaincantes, les Services acceptèrent même de passer la maîtrise d'oeuvre des systèmes (système d'armes compris) au constructeur (ex. du MIRAGE III-C, du MIRAGE III-E et surtout du MIRAGE IV).

- Souci de l'information du constructeur -

Le contact a été pris en général dès la création des programmes. Dans la plupart des cas le constructeur a été invité à faire une étude de faisabilité avant que le programme soit figé.

. Le constructeur sera toujours à même de suivre la vie de son matériel sur les bases aériennes et sera tenu régulièrement au courant des difficultés rencontrées tant sur le plan de la maintenance que sur le plan opérationnel.

- Souci d'une bonne politique en matière d'armement -

. Les choix officiels de base se sont révélés excellents (canon de 30 et rockets légères de 68). On a toujours de plus encouragé les efforts faits par l'avionneur pour améliorer l'emport de l'armement : c'est ainsi que pour notre part nous réalisâmes entre autres :

- combinés rockets-réservoirs (montés en pod)
- réservoirs porte-bombes (monté en pod)
- canon de 30 et munitions (montés en pod)
- containers contremesures ... etc...

1.2 - Sur le plan du constructeur, on peut porter à notre crédit :

- Souci de la qualité des équipes tant dans les bureaux d'études que dans les ateliers -

. Recrutement de qualité (fort pourcentage d'ingénieurs de grandes écoles).

- Souci de l'efficacité et de la rapidité -

. Les A.M.D. ont regroupé près de l'usine prototype les disciplines diverses associées dans la création de l'avion. Nous avons rassemblé à SAINT-CLOUD, les départements études avion - études servomécanisme - électronique E.M.D.

. Nous n'avons pas hésité quand une création ultra urgente s'imposait à donner un statut spécial à l'équipe appelée à réaliser le prototype (atelier particulier, personnel sélectionné), ceci pour aboutir dans les meilleurs délais.

- Souci de l'apport constant d'améliorations -

. Ceci suppose la présence de services de recherches actifs, toujours animés du désir de création que ce soient recherches sur procédés nouveaux, sur matériaux nouveaux ... ou sur formes nouvelles (service dit "d'aérodynamique théorique").

Dès que la mise au point d'un procédé nouveau semble assurée, on s'efforce aux A.M.D. de le glisser dans le prototype en cours.

- Souci de l'excellence des équipes appelées à établir les documents exigés pour la mise en série -

. C'était une condition essentielle pour travailler en bonne harmonie et efficacité avec les autres Sociétés tant françaises qu'étrangères associées à la production de nos avions - (diffusion à l'étranger des Mystère, Mirage, Falcon).

II - LE CHOIX DU MATERIEL - (Plusieurs cas) -

2.1 - L'état cherche son choix dans la compétition -

Tout part du programme. Les programmes successivement émis furent entre autres :

1) - Programme du chasseur polyvalent subsonique (intercepteur à vue - appui) -

. Ce fut le premier programme d'après-guerre. Ce programme était destiné à donner un sucesseur à notre production sous licence (VAMPIRE - MISTRAL). Deux avions se trouvèrent en concurrence.

- l'ESPADON de la S.N.C.A.S.O. (aile en flèche ~ 30°),
- l'OURAGAN des A.M.D. (aile à flèche très modérée).

Ces deux avions étaient équipés du réacteur ROLLS-ROYCE "NENE". L'alimentation en air du réacteur de l'ESPADON ne donnant pas entière satisfaction, l'OURAGAN qui avait une entrée frontale se révèla vite excellent et fut retenu. Il avait très sensiblement les performances du Shooting Star.

2) - Programme de l'intercepteur léger Mach 2 - (haute altitude)

Ce programme fut lancé par notre D.T.I. en l'accompagnant au départ de deux exigences :

a) - Avion équipé de deux réacteurs légers dotés de P.C. et en addition d'un groupe fusée utilisant acide nitrique + kérosène.

b) - Avion miniaturisé au maximum (influence des réalisations G N A T en GRANDE-BRETAGNE et A 4 D aux U.S.A.).

. Les deux appareils en compétition au départ furent :

- le S N C A S O - TRIDENT (biréacteur + fusée) - Aile mince et droite,

- le DASSAULT 550 (biréacteur + fusée) - Aile delta pur.

Le réacteur envisagé par la D.T.I. était au départ le TURBOMECA "GABIZO". La mise au point s'avèrant difficile, les A.M.D. proposèrent en remplacement le réacteur VIPER équipés d'une post-combustion étudiée et réalisée par A.M.D.

Un peu plus tard, la D.T.I., inquiète de sa politique moteurs, encouragea deux nouveaux concurrents :

- le S.N.C.A.S.E. DURANDAL aile delta,
- le S.N.C.A.N. GRIFFON aile delta.

Ces deux dernières réalisations s'écartaient du programme initial - d'une part, elles étaient toutes deux du type monoréacteur assisté par P.C. et groupe fusée dans le 1er cas et d'un pseudo-RAM-JET dans le second cas ; d'autre part, poids et taille avaient augmenté - Le réacteur était l'ATAR 8 de poussée statique 4,5 t.

Ce que voyant, les A.M.D. décidèrent de réaliser en un temps record et en P.V. le MIRAGE 001, monoréacteur ATAR 9B + P.C. + groupe fusée - fuselage avec area rule" - aile delta (identique à celle du MD-550).

Cet avion qui fit son premier vol neuf mois après la décision de lancement emporta la compétition et ouvrit la voie à son successeur le chasseur polyvalent tout temps MIRAGE III-A ou C (réacteur ATAR 9 C - radar CYRANO). Cet avion eut le succès international que l'on connaît.

3) - Programme de l'avion léger d'appui tactique -

Ce programme fut lancé par le NATO. Il opposa en compétition internationale :

- le FIAT G 91,
- le BREGUET TAON
- le DASSAULT ETENDARD VI

Le FIAT fut retenu.

4) - Programme du chasseur polyvalent V.T.O. (Mach 2) -

Le choix se fit "sur papier". Les concurrents étaient au départ BREGUET, NORD-AVIATION, SUD-AVIATION et DASSAULT. Ces deux dernières sociétés s'associèrent un peu avant la décision autour du projet présenté par DASSAULT. C'est ainsi que furent commandés d'une part le MIRAGE V (un réacteur de propulsion P.W. TF-306 + 8 moteurs de sustentation ROLLS-ROYCE RB-162) d'autre part, l'avion expérimental qui devait le précéder savoir : le BALZAC V (réacteur de propulsion Orpheus + 8 réacteurs de sustentation ROLLS-RB-153).

2.2 - L'Etat accepte les développements proposés par le constructeur -

Ce fut le cas après les premiers succès de l'OURAGAN (succès auprès de notre Armée de l'Air, succès à l'Etranger, en Inde en particulier - 360 construits).

Nous proposâmes successivement les variantes suivantes aux flèches de voilure de plus en plus accentuées.

- MYSTERE II - Flèche voilure 28° -

a) - Variante monoplace MD-452 -

. Aile à épaisseur relative 9 %. Empennage horizontal mobile - Gauchissement assisté par servo. Ce fut le premier avion européen qui franchit le mur du son.

b) - Variante bi-place radar MD-453 -

. Aile un peu plus mince en bout 8 % pour éliminer le wing dropping.

Les MYSTERE II étaient équipés d'un réacteur ROLLS-ROYCE "NENE". Deux exemplaires furent à titre expérimental équipés du réacteur ATAR à compresseur axial.

- MYSTERE IV - Flèche voilure 35° -

. Aile à épaisseur relative (7,5 %)

a) - Variante monoplace -

. Réacteur à compresseur centrifuge TAY puis VERDON.

b) - Variante biplace radar (dite IV-N) -

. Réacteur ROLLS-ROYCE AVON R A 7 (avec P.C.).

Le MYSTERE IV fut réalisé en série sous la variante MYSTERE IV (225 avions pour le compte du NATO, 375 pour le compte de l'Armée de l'Air française et pour l'Etranger).

- SUPER-MYSTERE SM.B1 et B2 - Flèche -

. Voilure 45° - Epaisseur relative 6 % -

a) - Variante monoplace réacteur AVON R 47 puis ATAR - 180 construits -

b) - Pas de variante biplace -

Pour que l'Etat acceptat de se laisser ainsi "mener" par le constructeur, il fallait que celui-ci ait de bons atouts en main.

Par exemple, quand nous proposâmes le MYSTERE II, nous pouvions faire état de notre expérience en matière de plans fixes réglables et en matière de servo-commandes. Ces éléments étaient visibles dans nos ateliers.

De même quand nous présentâmes la variante tout temps du MYSTERE II (MD-453), nous nous référions à l'expérience acquise en vol sur les entrées d'air latérales de l'OURAGAN "version photo" etc... etc...

2.3 - Le constructeur force le choix par une private venture réussie -

- Tel fut le cas quand nous eûmes la conscience de perdre notre position de force au profit de nos concurrents.

- M. Marcel DASSAULT prit ainsi en plusieurs occasions la décision de lancer des prototypes sans contrat.

- Le MIRAGE 001 fut ainsi réalisé (je l'ai déjà dit) quand nous eûmes conscience du danger que représentait pour nous DURANDAL, de la S.N.C.A.S.E., delta monoréacteur ATAR qui était en fait l'avion que nous aurions ainsi fait au départ si nous avions pu obtenir autre chose qu'un contrat pour un biréacteur VIPER P.C. (en attendant les GABIZO P.C.).

Le MIRAGE 001 s'imposa dès ses premiers vols (M = 1,5 dès le 5ème vol) et surpassa vite son concurrent. Nous gagnâmes ainsi grâce à la P.V. la compétition de l'intercepteur bisonique.

- L'ETENDARD 01 fut également réalisé en P.V., quand nous comprîmes que l'Armée de l'Air s'orientait vers l'utilisation de terrains plus modestes (BAROUDER). Nos efforts portèrent vers un développement de l'hypersustentation (becs basculants et volets à double fentes sur aile q = 45°). Cette initiative fut récompensée quand notre Marine Nationale adopta l'ETENDARD IV pour remplacer l'AQUILON sur ses nouveaux porte-avions.

- Le MIRAGE F 1 fut également réalisé en P.V. pour nous permettre de garder une position forte à l'exportation en créant ainsi un digne successeur du MIRAGE III.

Dans la même ligne de conduite, toujours en P.V., il y a lieu de citer le MIRAGE à moustaches dit MILAN, développé en liaison avec la Direction Technique de l'Usine Fédérale Suisse d'EMMEN.

Cette initiative a essentiellement pour but de fortifier la position du MIRAGE face à la concurrence. Elle aura très certainement des retombées heureuses sur nos avions "chauds" futurs et sur CONCORDE.

- Le MIRAGE G (Géométrie variable) -

fut au démarrage lancé sans les crédits de l'Etat qui, au départ, voulair se limiter à des études et des réalisations partielles (telles que pivot) et qui ne se décida à financer les finitions que bien après le lancement du prototype dans nos ateliers.

- Le MYSTERE 10 (Mini-Falcon) - Le MYSTERE 20 (Falcon), sont nés par le fait de la P.V.

Tels sont également les cas du COMMUNAUTE et de l'HIRONDELLE.

- C O N C L U S I O N -

En résumé, les Avions Marcel Dassault dans ces vingt dernières années ont créé en moyenne chaque année un prototype d'un modèle foncièrement nouveau.

Les commandes de série ont porté sur :

- 360 OURAGAN,
- 600 MYSTERE,
- 180 SUPER-MYSTERE
- 100 ETENDARD IV,
- 1000 MIRAGE III,
- 62 MIRAGE IV.

La production de ces avions, réalisée en partie par les autres Sociétés Nationales ou privées, a entraîné en moyenne la participation d'environ 30.000 personnes soit sensiblement 1/3 de l'effectif total de l'industrie aéronautique française.

Ces avions ont, de plus, fait l'objet de contrats auprès de nombreux pays étrangers notamment : Inde, Israël, Australie, Afrique du Sud, Suisse, Belgique etc... assurant aux A.M.D. une position de tout premier plan parmi les premiers exportateurs français.

Tel est ce bilan très favorable qui démontre bien que la meilleure méthode était de "Challenge for Survival".

THE USE OF TRADE-OFF STUDIES IN PRELIMINARY DESIGN

by

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THE USE OF TRADE-OFF STUDIES IN PRELIMINARY DESIGN

Helmut Langfelder

1. SUMMARY

Preliminary design of new combat aircraft is not usually based on completely defined specifications. Fixing the major design parameters or making salient design decisions to arrive at the "optimum configuration" and developing a sensible basic specification for the weapon system are processes which mutually influence each other. Furthermore, this period of conceptual study takes so long a time that there are distinct and sometimes fundamental changes in the views of the user of how the aircraft will best be employed to meet presumed needs. The only constant factor seems to be the necessity to keep costs and complexity to a minimum, the idea of "cost-effectiveness". Though there is a broad consensus of opinion of what is required of a combat aircraft in qualitative terms, there is no unanimity as to how valuable the various factors are in relation to each other. Any attempt to trade off quantitatively alternative design features or parameters must proceed within the total weapon system concept and should yield better insight into the nature of optimum mix. Isolated emphasis on any single factor, such as manoeuvrability, range, payload, armour protection and offensive armament, electronic sub-system capability, maintainability or survivability would be misleading. This technique of balanced effectiveness studies is illustrated by a discussion of such questions as wing and thrust loading choices, sizing, speed capability, number of engines, aerodynamic configuration, aircraft sub-systems, automatic check-out and electronic system complexity as seen in relation to weight and cost penalties.

2. INTRODUCTION

Trade-off studies are the generally accepted method of quantifying the multitude of preliminary design decisions. In particular, the combat aircraft weapon system can only be represented by a multi-dimensional complex of parameters, some of which are traditionally well-understood but many others have a strong emotional content. There is no infallible technique of clearly demonstrating the inter-relation of design decisions and the fulfillment of specification aims in many important instances. The problem is basically one of simultaneously satisfying very different conditions and setting priorities in a flexible design which also meets many off-design requirements. It is the purpose of this paper to prompt discussion of this problem within the frame-work of the weapon system approach. Some recent attempts will be indicated to present analysis in such a manner that a more rapid convergence of a widely acceptable specification of performance and design requirements on the one hand and a feasible engineering solution on the other is achieved. During the initial design process, a change of views on the requirements frequently takes place. For example, the feasibility of successful air combat and the effectiveness of missile or other armament, such as guns, the reliance on electronic aids or the cost-effectiveness of multi-purpose designs in a wide performance spectrum have been assessed in very different ways in the past.

3. SOME FUNDAMENTAL ASPECTS OF TRADE-OFF ANALYSIS

As in every complex problem, it is important to ask the right questions. The type of study that can be undertaken varies from using a basic configuration and making limited excursions to a general application of stretched technology. One must decide correctly exactly how far to go. The limitations of analytical study should be appreciated and wherever possible tests and hardware must support the analysis. During Feasibility or Conceptual Study periods many aspects are really only qualitatively introduced in the wide survey. Such studies yield tentative answers, the quantitative significance of which is often over-estimated. The subsequent Definition Phase must result, however, in very reliable proposals. Clearly, in that case the trade-off study must be made to apply only to relatively small variations on a well-defined baseline configuration. Precise and significant answers in that phase, of course, are even more indispensable whenever some type of total package procurement is expected to result. Whether this is realistic or not depends largely on how successfully trade-off studies are applied.

An interesting question is how this study process can be subjected to schematization. This has been attempted in the system engineering process as examplified by the US Air Force 375-series of regulations. A rigorous derivation of design requirements by trade-off analysis starting from functional flows is, of course, an ideal never to be achieved, only at best to be approached. One difficulty resides in the fact that the assumptions underlying the functional flows must be themselves be affected by the results of the trade-off studies. The problem is indeed too complex even for very sophisticated recipes. Good examples of the complexity of the problem can be drawn from avionics and in particular from analyses of kill probabilities of systems using advanced target acquisition and weapon delivery aids. Numerous factors from the whole range of aircraft technology enter into this problem. All aspects of the basic performance of the airframe, the functioning of the man-machine relationship of the pilot reacting to displays and other cues in the real situation, the reliability and accuracy of the electronic aids and the weapon itself must all be considered in relation to the target environment within practical technology and cost limitations. Without attempting to approach anywhere near answering such ultimate questions, which can have no simple quantifiable answer, it is still profitable to consider realistically restricted basic design parameters by trading the accessible and known effects against each other.

4. BASIC DESIGN DECISIONS

The specification of requirements enabling tentative basic design decisions will apply to traditional aircraft parameters. A summary of such quantities is shown on Figure(1) as applied to a combat aircraft for a given state-of-the-art in aerodynamics, propulsion, and structural and equipment technology, i.e. weight. The initial trade-off studies will apply to the sizing of the weapon system, i.e. a definition of the design weight and the thrust and wing loading. Many possible choices can be made in specifying the relevant performance parameters with varying emphasis on range, maximum speed, take-off and landing and manoeuvrability parameters. In each group those parameters are shown which have been found to be the necessary minimum for a satisfactory definition of the problem. Each of these fifteen parameters must be regarded as a function of practically all the others. The effect of the various parameters on size, wing, and thrust loading is clearly different. Range will predominantly affect size and the manoeuvrability mainly either wing or thrust loading, but some, such as specific excess power, will also strongly affect sizing via the required engine size to provide the necessary thrust loading.

Before it is possible to display the typical influence of some of these parameters on the basic design, some thought should be given to the region of performance optimization as shown on Figure(2). The altitude - Mach number diagram includes the P_s-contours which at any given sizing are strongly dependant on the region of optimization, which in the example shown is for a typical combat aircraft optimized for low-level operation, i.e. the main performance and range requirements are in the subsonic speed range at altitudes below 10,000 ft. Immediately, the question of the trade-off aspects of an extension of the performance envelope to higher supersonic speeds for a secondary performance capability suggests themselves. The effect of the air inlet in this connection will be discussed later.

Whereas the sizing influence of range requirements is a standard subject for such parametric studies, the manoeuvrability requirements, in particular, have only more recently been fully incorporated in a systematic way. Figure (3) shows a plot of wing loading and thrust loading for a fixed configuration and propulsion system type for a given range/payload design as affected by some of these parameters. Typically, the energy manoeuvrability and the touch-down speed requirements define the required relationship of wing loading and thrust loading. The high values of the latter are characteristic of the present situation of combat aircraft design. Such plots can also be used to compare different configurations, for example variable and fixed geometry wings or fan and straight jet engines but since the designing requirement will then most likely change, the need immediately arises of a more comprehensive plot of a larger number of parameters, including the design weight. The size of aircraft required to meet all the stated requirements is often regarded as a basic yardstick of optimization, though its real functional relationship with costs is not precisely known.

A type of carpet plot for the design requirements trade-off which has proven very useful and can be applied to a wide variety of parameters is exemplified by the graphs of Figure (4). While this still applies to a fixed configuration and engine type, it enables cross-comparison of the effect of making the design simultaneously meet many requirements and also assesses the relative changes of the parameters. Using a specified mission profile with prescribed payload, both thrust loading and weight is varied, the latter by increasing the fuel carried with fixed airframe size. A mission radius of 200 nm., for instance, is possible at a thrust loading of 1.0 with a nominal overload factor of about 15% which gives a specific power of just under 600 f.p.s., a turn rate of 13^{o}/sec, a touch-down speed of 107 knots and a take-off ground roll of 1350 ft. The penalty to be paid by increasing range by overload or specific power by increased thrust loading in terms of the other requirements becomes apparent.

The significance of such results is very dependant on how carefully configuration design is treated. Clearly large variations in size or thrust loading with resulting changes in the aircraft geometry and balance must be studied by a sufficient number of detailed point designs with weight assessment. Assuming this is done, the general survey of parameters should result in an isolation of essential characteristics which will predominantly influence the design decision and, in particular, will approach the problem of the "optimum" configuration. A rational choice of this in broad respects, such as design primarily for range and penetration, or manoeuvrability is essential. Figure (5), for example, attempts to present the manoeuvrability characteristics of a combat aircraft in low level flight on a plot of attainable load factors as a function of Mach number. This cannot be specified by any single value, such as one of the three parameters defining the manoeuvrability at low speed as turn radius, high speed, high g as turn rate or the high speed, one g value of the energy manoeuvrability. Using a fixed configuration, low speed instantaneous g-values are shown up to the structural limit on the left hand boundary. Thrust limitations with afterburner and military thrust are indicated for higher speeds. In this part of the diagram lines of constant attainable P_s are included, with decreasing specific power up the maximum constant load factor, where $P_s = 0$, and increasing specific power to its maximum value in the transonic region for non-manoeuvring flight. The buffet penetration used to limit the useful g attained must ensure a satisfactory operating platform for combat flight. This plot can be used to compare various configurations to select one for superior combat qualities. Though it cannot be expected that any one aircraft is at an advantage in all respects, it is useful to know where it excels to exploit its advantages in combat tactics. This applies particularly to variable geometry aircraft which offer considerable flexibility by being capable of adjusting the configuration to flight requirements. Here it may be desirable to indicate manoeuvrability potential to the pilot by continuously displaying a parameters, such as P_s for example.

5. PROPULSION

There is a growing tendency to prefer twin engine installations for combat aircraft. When the high thrust requirements enforce such a decision because available engines are not big enough for a single engine design, the question of a detail trade-off analysis as to the more cost-effective course of action may not arise. It is, however, of considerable interest to study the problem without the restriction of engine availability and to attempt to justify, if possible, the twin-engine design on a cost-effectiveness basis. This is generally attempted by an analysis of total cost of operation. The twin engine design is heavier and this is normally accepted. Figure (6) shows the weight distribution of comparable single and twin-engined combat aircraft designs. The empty weight penalty is seen to be 8,8%. This is due to the fact that the engine installation itself is somewhat heavier, but also in large part because the airframe structure grows and there are additional problems with the afterbody design for low transonic drag. The problem of balance of the configuration is accentuated, especially at high installed thrust because of the bigger weight in the rear. All this escalates the fly-away-cost of the aircraft, which is further increased by the fact that the cost per pound of thrust is bigger with smaller engine units. The case for the cost-effectiveness of twin engine combat aircraft is thus seen to rest entirely on assumed lower loss rates in peace-time operation or better survivability in war-time action. Available statistics can be used to make such a case. It will all depend on how these statistics are interpreted to predict probable future experience with the new combat aircraft design.

One of the problems of twin engine installations has already been mentioned. It is more difficult to optimize the afterbody for transonic operation both with and without afterburner, especially in the case of a secondary high supersonic requirement. A study of the value of fairings between the engines and optimum engine spacing is shown on Figure (7). Drag reduction can be achieved by such a fairing which may either be a small fixed fairing to accommodate the opened nozzle or a variable one which can eliminate most of the harmful base area but will involve more weight. The particular case studied indicates that up to 9% of the complete aircraft drag can be eliminated, and this for a particular mission profile would be equivalent to 130 lb available for the fairing, taking into account the re-balance of the whole design associated with additional weight in the rear location. The estimated weight for such a fairing is only 80 lb, so that it appears that the added complication would pay. The drag reduction also benefits such parameters as energy manoeuvrability specified in the transonic regime, but, of course, does not necessarily improve low speed manoeuvrability. Further points that must be considered are maintainability and reliability.

An interesting example of a general inlet and exhaust system trade-study applied to a variety of assessment parameters including weight, performance, cost, and maintainability is shown on Figure (8). This had been carried out as part of an investigation to study the effects of extension of the whole operating envelope of a combat aircraft to higher Mach numbers at altitude. Such an envelope was shown on Figure (2), where the inlet and exhaust system effectively restricted the maximum Mach number to about 1.6 in spite of high thrust loading, which basically would enable the aircraft to reach much higher Mach numbers. But this entails more complex inlets and nozzles with associated penalties. In this case five different types of inlet and nozzle combinations were studied ranging from a very simple inlet with rounded lips and no moving parts except an auxiliary door for take-off operation to a fully variable external compression inlet and convergent-divergent nozzle. The simple inlet can be optimized for the transonic performance requirements of the aircraft and is, therefore, taken as reference. It is seen that the supersonic inlet installation results in appreciable degradation of most of the assessment parameters. The Mach number can be raised to normal airframe continuous operating limit, but the design take-off weight can increase by nearly 10%, unless a fully variable inlet is used to restore some of the transonic operating efficiency. Even in this case nearly 6% increase in take-off weight or 8% decrease of mission radius will have to be accepted. Other transonic manoeuvrability parameters, such as P_s or time to accelerate to $M = 0.9$ similarly suffer degradations of nearly 9%. The cost of development and production of the air inlet and exhaust system are more than doubled, and there is a very considerable increase of the maintenance effort required by the more complex installation. This example demonstrates the severe effect of certain secondary performance requirements, which will prevent any true optimization within the performance envelope and will lead to rapid cost escalation. The results of this study are interpreted to indicate that only a definite mission requirement in the supersonic regime justifies operational capability at Mach numbers in excess of about 1.6.

6. DOES COMPLEXITY PAY OFF ?

Amongst the most difficult decisions in preliminary design of a combat aircraft are the many questions relating to the specifications of equipment which may be "non-essential". The most powerful way of saving weight and cost is to exercise some restraint in this field. As Figure (9) shows, about half the fly-away-cost of the aircraft is attributed to equipment of some sort. The weight of these items is typically about 20%, but can be much more for very complex weapon systems. While variable airframe features, such as high lift or manoeuvre devices, variable sweep, variable inlet and nozzles, etc. can be assessed in their cost effectiveness against the performance requirements as has been discussed, the direct influence of sophisticated and complex sub-systems appears as cumulative weight increases and increased complexity, which must be justified by assessing their true utility. The attitude towards such improved capabilities varies from specifying everything that is costly and complex to excessive austerity. The aim is to decrease pilot work load by automation, improve maintainability by automatic check-out and monitor systems or reliability by redundancy. It is not that complexity itself must be avoided at all cost, but rather that especially in military equipment it should never be introduced without careful justification. Our analysis techniques of cost-effectiveness of many equipment items is in a very rudimentary state. The emotional content is considerable. One must guard against the application of technologically interesting features for their own sake and the user's natural tendency to specify increased capability if it appears at all attainable.

Particularly in the case of the electronic sub-system a balanced approach is essential. Figure (10) attempts to summarize the problem of ensuring that improved capability penalties are justly balanced against increased operational utility. The tactical employment of combat aircraft is not well understood in theoretical analysis. It is very difficult to quantify increased mission effectiveness of such capabilities as are provided by bigger air search or target acquisition radars, automatic attack procedures or all-weather precision ground attack systems. On the penalty side not only the obvious degradation of flight performance and increased cost but also the effect on maintenance effort and availability has to be assessed. The latter is a very important consideration.

Somewhat more accessible is the question of optimum passive protection. This is considered under the dual headings of fuel protection and armour. Figure (11) sets out the results of a vulnerability analysis, which groups certain protective features as "imperative", "recommended" or purely "desirable". Vulnerability due to fire and explosion represents a prime hazard for the combat aircraft in operations exposed to small arms fire. Protective armour is also required in such an environment. The best protection of the aircraft may be superior performance and its ability to escape the hostile environment. This is the negative aspect of heavy protection, which adversely affects the performance. By an analysis of vulnerable areas of a combat aircraft in low-level penetration flight the probability of its destruction is calculated for an assumed scenario. This leads to the classification of the items shown in relation to their weight penalty expressed as percentage of the aircraft empty weight. In the case considered 2.1% of weight was found to be "imperative", mainly to be used for explosion suppression. Armour protection of crew and critical components is "recommended". The case considered was for a two-man crew, and in this case interstation crew protection to eliminate total loss by single hits in the cockpit is imperative. A total weight of about 5 1/2% represents a considerable investment in passive protection.

7. CONCLUSIONS AND RECOMMENDATIONS

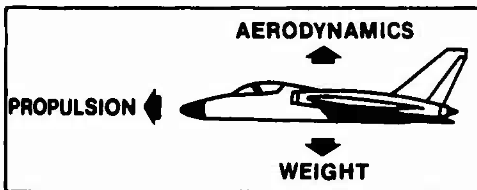
In this short survey many important questions have not been touched on at all. For example, the strength of airframe and the design life of the combat aircraft should also be investigated under the trade-off aspect. The redundancy of structure and systems has to be looked at for the case of the battle-damaged aircraft for safe return. But the examples cited are sufficient to illustrate some main conclusions, which are imperative in the present situation of combat aircraft design.

(a) Trade-off studies on a wide range of parameters are the only means of rationalizing basic design decisions, which make or mar the aircraft weapon system as an effective military instrument.

(b) Such analysis will provide some insight into the correct application of new technologies, which must be critically examined before adopting them for a successful aircraft.

(c) The possibilities of a multi-role design have to be assessed by trade-off studies to compare it with designs for a more restricted performance spectrum.

(d) The preliminary design techniques in many areas are still in their infancy. Much effort is justified in improving them to develop satisfactory combat aircraft specifications without courting disaster by misjudging time scale, development risk or having cost getting completely out of control.



	DESIGN WEIGHT (SIZING) THRUST LOADING WING LOADING
RANGE PARAMETERS	PAYLOAD / ALTITUDE / SPEED COMBAT RESERVES
MAX. SPEED PARAMETERS	MAXIMUM MACH NUMBER SEA LEVEL MAXIMUM MACH NUMBER 36000'
TAKE-OFF AND LANDING PARAMETERS	TAKE-OFF DISTANCE LANDING GROUND ROLL TOUCH-DOWN SPEED
MANEUVERABILITY PARAMETERS	LOAD FACTOR TURN RADIUS TURN RATE ACCELERATION SPECIFIC EXCESS POWER P_s

Fig. 1 Basic Design Decisions

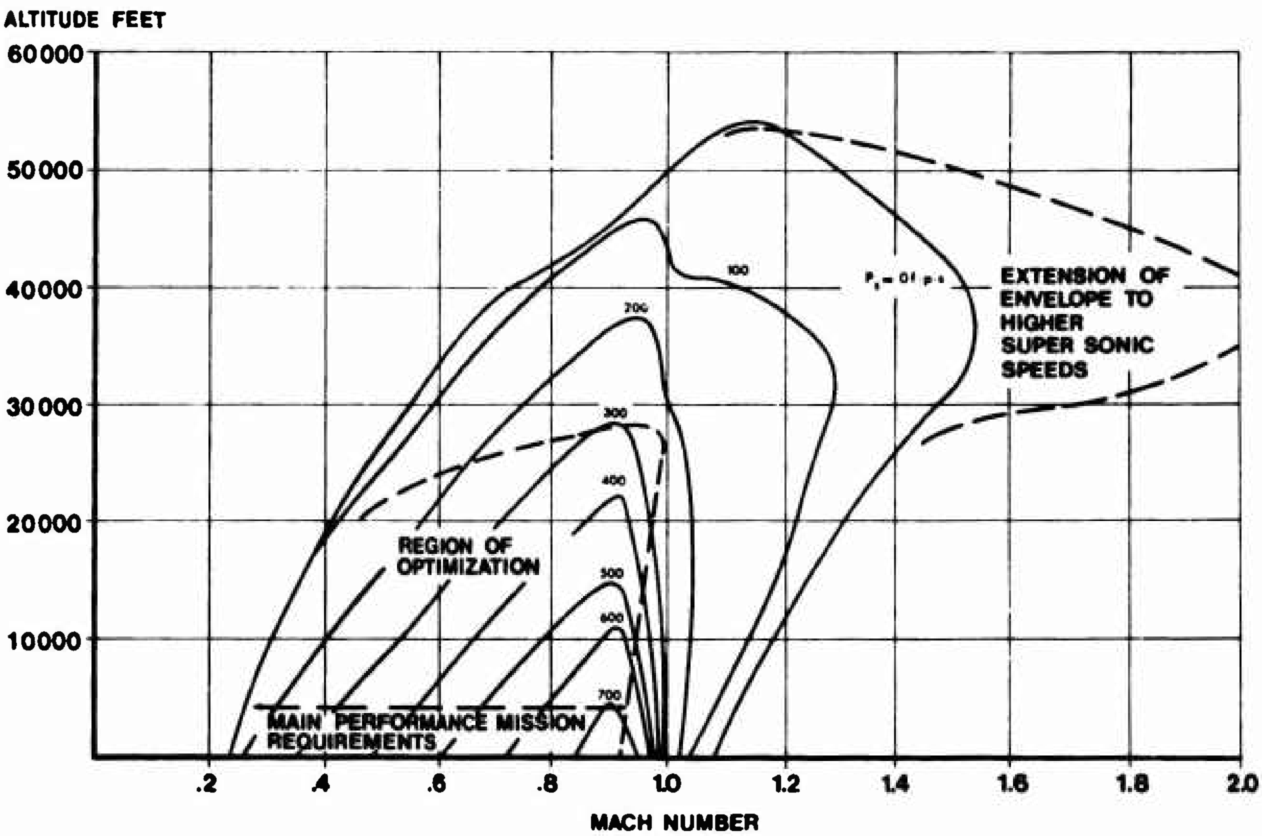


Fig. 2 Combat Aircraft Performance Optimization

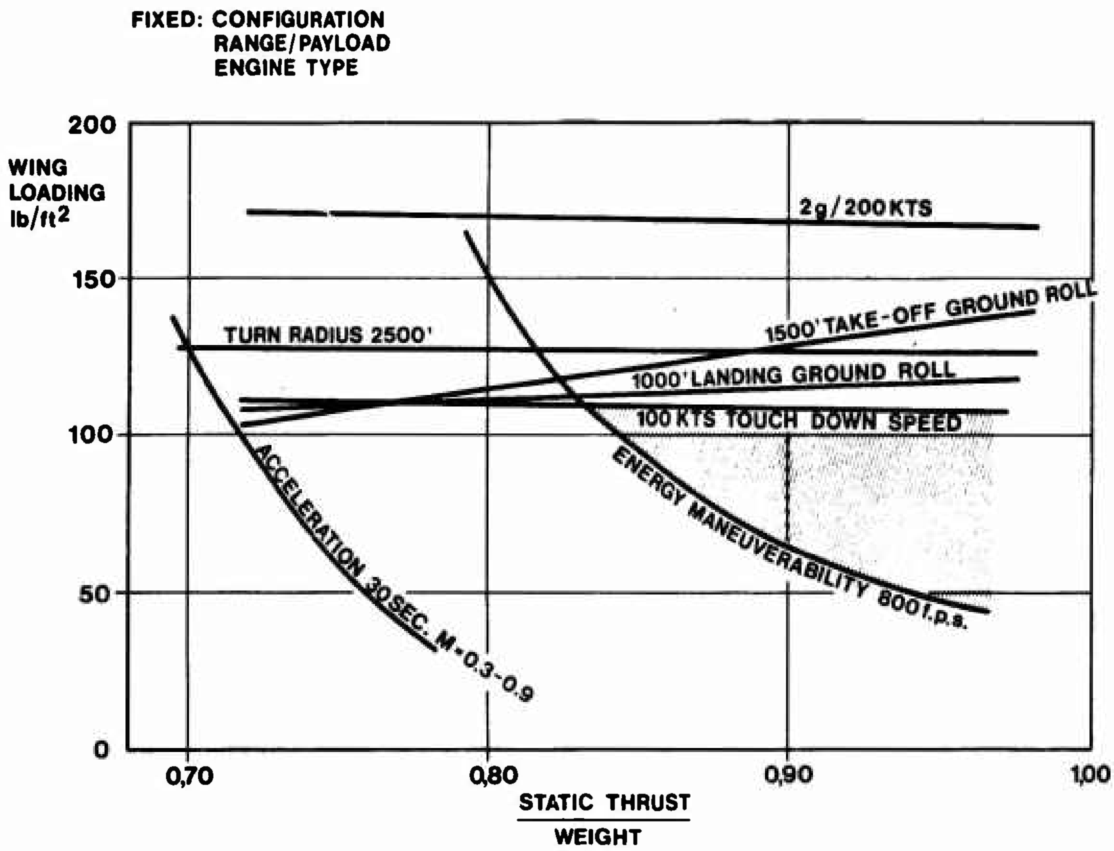


Fig. 3 Choice of Wing Loading and Thrust Loading

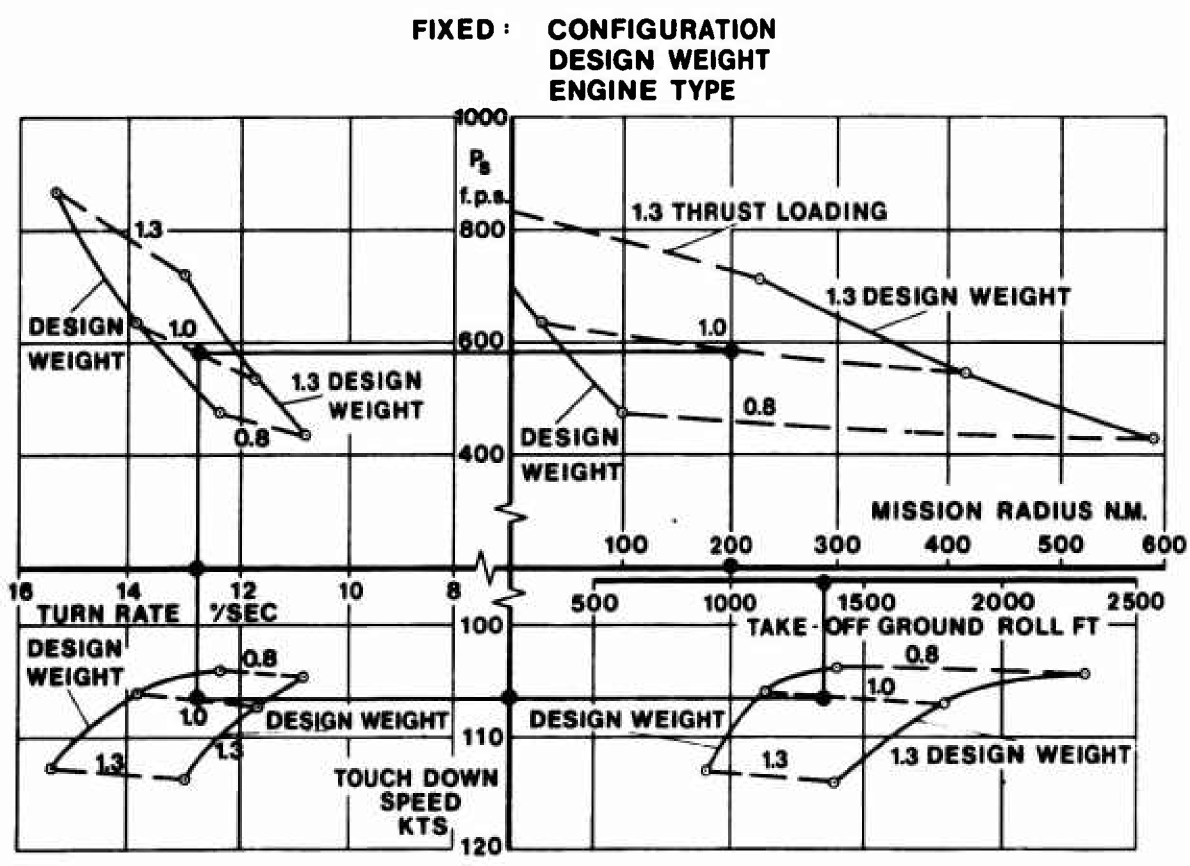


Fig. 4 Design Requirements Trade-off

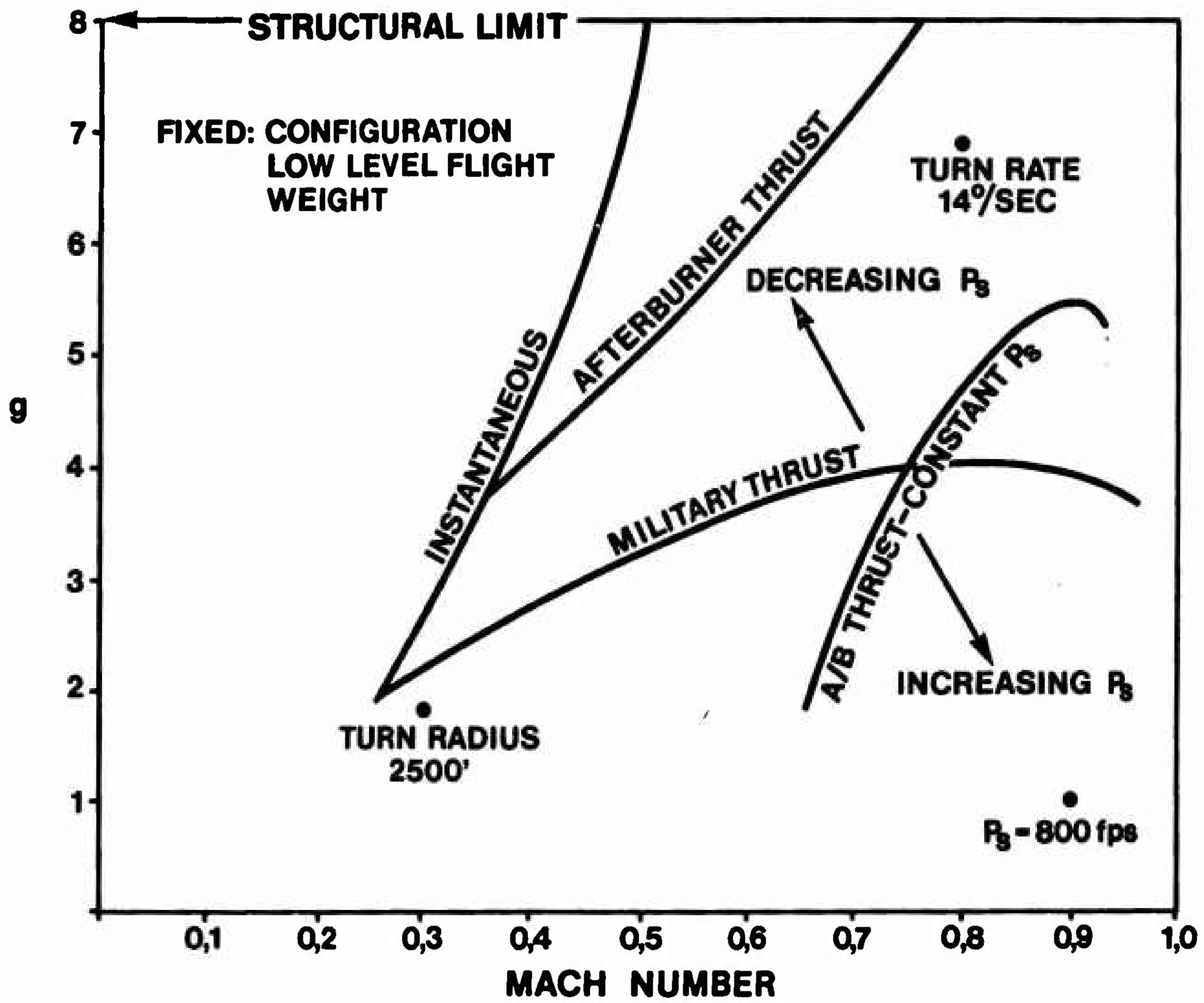
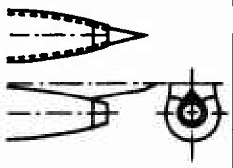
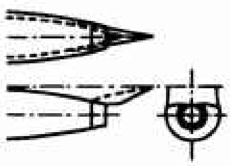


Fig. 5 Manoeuvrability Characteristics of Combat Aircraft

	SINGLE ENGINE	TWIN ENGINE
STRUCTURE	50,2 %	54,6 %
PROPULSION	30,2 %	33,5 %
EQUIPMENT	19,6 %	20,7 %
EMPTY WEIGHT	100 %	108,8 %
USEFUL LOAD	24,2 %	24,2 %
MISSION FUEL	34,0 %	36,8 %

Fig. 6 Weight Distribution for Single and Twin-Engined Combat Aircraft

NO FAIRING BETWEEN ENGINES LARGE BASE AREA		DRAG REDUCTION REFERENCE CASE	EQUIVALENT WEIGHT AVAILABLE FOR FAIRING	ESTIMATED WEIGHT OF FAIRING
FIXED FAIRING TO ACCOMODATE AFTERBURNER JETS SMALL BASE AREA		DRAG REDUCTION 5%	70 LB	50 LB
VARIABLE FAIRING OPTIMIZED FOR MILITARY THRUST NO BASE AREA		DRAG REDUCTION 9%	130 LB	80 LB

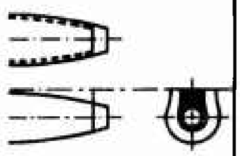


Fig. 7 Twin-Engine Installation Optimized for Transonic Speeds

INLET AND EXHAUST SYSTEM TRADES

TYPE OF INLET AND EXHAUST INSTALLATION	SIMPLE INLET ROUNDED LIPS NO MOVABLE PARTS AUXILIARY DOOR OPERATED BY SWITCH IN COCKPIT CONVERGENT VARIABLE NOZZLE	HALF CONE SHARP LIPS NO BY - PASS BUZZ CONTROL PRESSURE SENSOR AUXILIARY DOOR CONVERGENT VARIABLE NOZZLE	HALF CONE SHARP LIPS WITH BY - PASS DIFFUSOR MACH NUMBER SENSOR AUXILIARY DOOR CON-DI NOZZLE	TWO-DIMENSIONAL DOUBLE RAMP FIXED CAPTURE AREA VARIABLE SECOND RAMP LOCAL MACH NUMBER PROBES AUXILIARY DOOR CON-DI NOZZLE	TWO-DIMENSIONAL DOUBLE RAMP VARIABLE CAPTURE AREA VARIABLE SECOND RAMP MACHNUMBER AND PRESSURE SENSORS AUXILIARY DOOR CON-DI NOZZLE
RELATIVE DESIGN MISSION RADIUS LO-LO-LO, HIGH SUBSONIC	1.000	0.951	0.896	0.875	0.920
RELATIVE DESIGN TAKE-OFF WEIGHT	1.000	1.046	1.075	1.098	1.058
MAX. MACH NUMBER 36 000'	1.60	1.90	2.14	2.30	2.30
RELATIVE SPECIFIC FXCESS POWER P_S AT M = 0.9	1.000	0.942	0.916	0.929	0.934
RELATIVE TIME TO ACCELERATE M = 0.3 - 0.9	1.000	1.099	1.134	1.087	1.087
RELATIVE DEVELOPMENT COST (INLET/EXHAUST)	1.000	1.327	1.477	1.917	2.516
RELATIVE PRODUCTION COST (INLET/EXHAUST)	1.000	1.10[illegible]	1.627	2.016	2.099
RELATIVE MAINTEN..NCE EFFORT (INLET/EXHAUST)	1.0	1.5	9.7	11.0	37.0

Fig. 8 Inlet and Exhaust System Trades

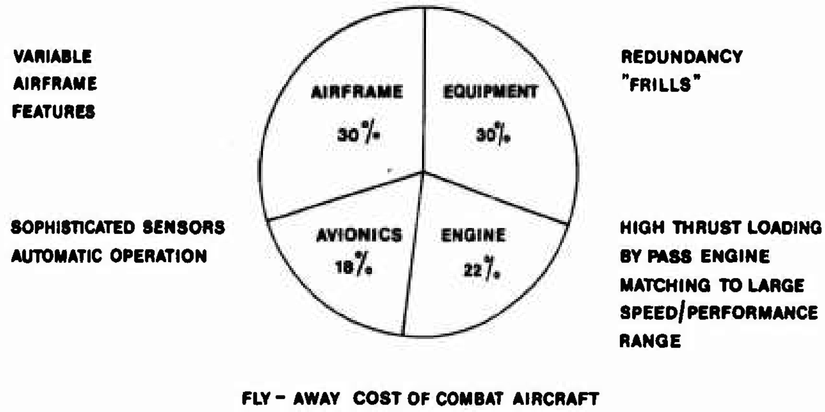


Fig. 9 Does Complexity Pay-off?

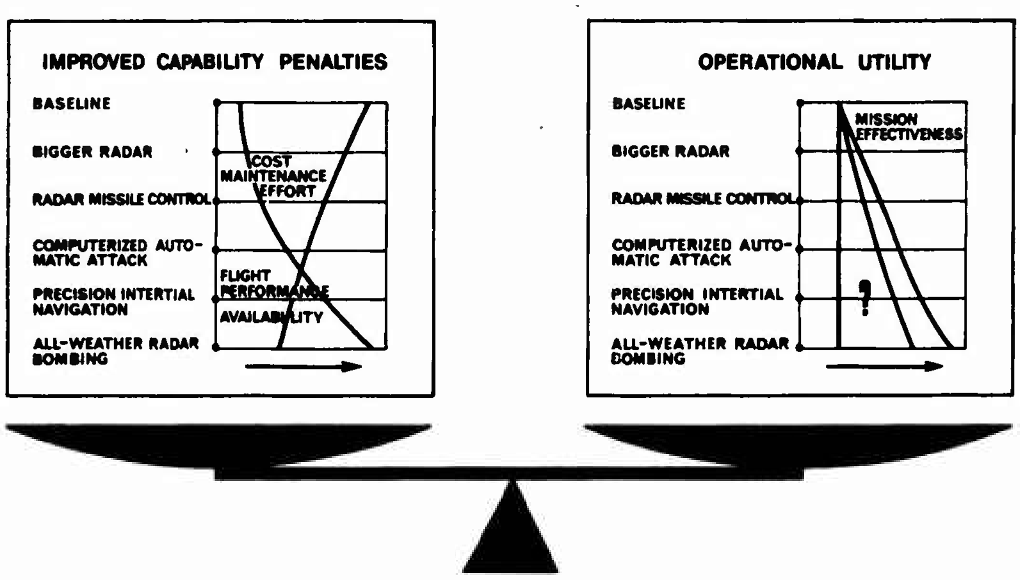


Fig.10 Balanced Avionics Approach

PERCENT OF AIRCRAFT EMPTY WEIGHT	IMPERATIVE	RECOMMENDED	DESIRABLE
FUEL PROTECTION			
INTERNAL FOAM IN FUEL TANKS	1.09	-	-
EXTERNAL VOID FILLER	0.24	-	-
SELF-SEALING OF FUEL CELLS AND FUEL LINES	0.67	0.34	0.67
ARMOUR			
INTERSTATION CREW PROTECTION	0.10	-	-
CRITICAL COMPONENTS-FLYING CONTROLS	-	0.60	-
CRITICAL COMPONENTS - FUEL VALVES	-	0.29	0.26
BULLET-RESISTANT WIND SHIELD	-	0.13	-
CREW ARMOUR ON FUSELAGE SIDE	-	-	1.08
TOTAL PROTECTION 5.47 %	2.10	1.36	2.01

Fig.11 Optimum Passive Protection

AEROELASTIC CONSTRAINTS FOR LARGE AIRPLANES WITH A CANARD CONTROL

by

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U.S.A.

SUMMARY

The stability of large, low load factor design, supersonic airplanes is significantly effected by aeroelasticity. Such airplanes require careful preliminary design analysis to insure that adequate stability and control is achieved in the final design without major redesign. The impact of aeroelastics on stability is particularly important for airplanes with a canard pitch control surface. The sensitivity of airplane stability, to weight, mass distribution flexibility, and aerodynamic characteristics must be considered in arriving at an adequate design. An example design is discussed.

NOMENCLATURE

ac	Aerodynamic Center
ac_R	Rigid Airplane Aerodynamic Center
ac_E	Elastic Airplane Aerodynamic Center
C_R	Root Chord
G.W.	Gross Weight
V_{DIVE}	Limit Dive Velocity
V_{MO}	Limit Maximum Operating Velocity
$V_{DESCENT}$	Descent Velocity
M	Mach Number
LB.	Pounds

INTRODUCTION

Large airplanes, especially large supersonic airplanes such as advanced bombers or transports, have large moments of inertia. As a result their control response is more sluggish than that of smaller airplanes. Slow response to pilot control inputs result in degraded flight path and attitude control which in turn effect an airplane's relative safety, especially during rough air approaches and landings. Relatively improved landing approach control results from the use of a canard for pitch control such that an increase in pitch attitude and wing lift is achieved by the addition of a lift force on the canard rather than a downward force on an aft tail or wing surface.

A canard control surface must be a carefully integrated part of the configuration in order to avoid or minimize undesirable flight characteristics, i.e.: (a) Adverse aerodynamic interference on the wing, the vertical tails, or the engine inlets; (b) increased roughness of ride for the crew; (c) aeroelastic stability problems. This paper deals with item (c) aeroelastic stability problems.

THE PROBLEM

Preliminary design studies were made of a large, supersonic transport type, variable sweep airplane with a canard pitch control (Fig. 1). The aerodynamic and dead weight load distributions along the fuselage wing, and canard for steady ($\eta = 1$) flight is roughly as indicated in Fig. 2. The summation of all the vertical loads and all the moments about the center of gravity must be zero.

When the fuselage is lightly loaded (no fuel or payload in the forward fuselage), Figure 3, a sudden change in angle of attack, like a gust, can add relatively more lift to the canard and forebody than to the wing because of the fuselage flexibility and the lack of forward fuselage inertia to resist the incremental lift. Such fuselage deflection and increased aerodynamic lift cause the effective aerodynamic center (for the dynamic, control fixed case) to move forward and is destabilizing. Conversely when the forward fuselage is heavily loaded its inertia tends to cause the fuselage to bend downward when a sudden upward gust hits the airplane, resulting in reduced aerodynamic lift on the canard and forebody (Fig. 4). In this case the effective dynamic aerodynamic center moves rearward and is stabilizing. It is necessary to predict the incremental aeroelastic stability effects with sufficient accuracy to be certain of adequate total airplane stability and control for all flight conditions.

Figure 5 presents a simplified equation for dynamic aerodynamic center location. It is comprised of the rigid airplane aerodynamic center location corrected for rigid canard effects, elastic airplane and canard effects (including inertia effects), and airplane and canard maneuver damping effects.

The elastic characteristics of the wing, fuselage, and canard are indicated by their respective stiffnesses such as Figure 6, the fuselage moment of inertia distribution. Similarly inertia effects are related to the mass or weight distribution.

The aerodynamic effect of a given deflection, or angle of attack, or camber change is proportional to the airplane flight dynamic pressure, Fig. 7. The maximum dynamic pressure occurs on the dive velocity placard, consequently the aeroelastic effects are the largest along the dive placard boundary. The aerodynamic characteristics of the wing, fuselage and canard must be estimated or derived from wind tunnel tests.

The difference between the calculated aerodynamic centers for the elastic and rigid cases is presented as the aeroelastic effect on aerodynamic center. Figure 8 indicates that the airplane without a canard has a maximum aeroelastic aerodynamic center variation of 6 percent of the root chord, at about .8 Mach number on the dive velocity placard. The higher gross weight condition, which has a heavily loaded fuselage, has smaller aeroelastic effects, especially at supersonic Mach numbers.

Figure 9 presents the incremental aerodynamic center due to aeroelastic effects for the complete airplane including the canard control plane. As in the canard off case, the higher gross weight reduced the net aeroelastic aerodynamic center variations. The largest variations occurred in the transonic speed regime and at the highest operating dynamic pressures (on the dive placard).

The difference between the canard on and canard off characteristics in Figure 10 shows the canard to contribute aeroelastic variations of up to 6 percent stabilizing for the light weight, lightly loaded fuselage at 1.2 Mach number on the dive velocity placard. Conversely the dynamic aeroelastic stability contribution of the canard was destabilizing at the high subsonic Mach numbers and highest supersonic Mach numbers for the high gross weight case.

Since the dynamic aeroelastic stability effects are large at Mach 1.2 and are a maximum at the highest flight dynamic pressure, a number of sensitivity studies are presented for the Mach 1.2, dive velocity placard condition in Figures 11 through 16.

Figure 11 indicates that a 25 percent change in canard lift curve slope results in only about one percent change in the aeroelastic stability contribution. The canard stiffness can have a large effect on the aeroelastic stability increment, especially for the high gross weight case (Fig. 12).

Any reduction in forebody stiffness can result in large increases in aeroelastic stability variations (Figure 13) for the light gross weight case.

Figure 14 indicates that fuselage forebody lift curve slope variations are not of great significance.

Since the dynamic aerodynamic center is a function of the fuselage inertia effects and the normal accelerations to which the fuselage is subjected, wing lift curve slope variations (which occur with planform changes) can significantly effect the aeroelastic stability increments as indicated in Figure 15.

In order to better understand the dynamic aeroelastic stability effects, the variations with flight dynamic pressure of the pure aeroelastic effect (no mass), the pure inertia relief effects and the net effect of combining these two are presented in Figure 16. The inertia relief effect is related to the normal acceleration that the airplane experiences upon encountering a gust and the fuselage deflection that results because of the mass distribution along the fuselage. A given angle of attack change causes a smaller normal acceleration for the high gross weight airplane than for the light gross weight airplane, consequently its inertia relief is less (provided the fuselage mass distribution is the same). The data on Figure 16 indicates that the inertia relief for the light weight case is 30 percent greater than for the heavy weight case. It is noteworthy that the ratio of gross weights would indicate a 45 percent greater normal acceleration. The relatively greater stabilizing effect of inertia relief for the 605,000 pound case is the result of the increased mass distribution along the fuselage for this heavy gross weight case. It is noteworthy that the 20 percent destabilizing contribution of the pure aeroelastic effect is more than offset by the inertia relief effect. The result is a net effect of less than 8 percent root chord shift in the dynamic aerodynamic center.

CONCLUSIONS

It is important that any preliminary design configuration be arranged so the combined effects of the rigid aircraft aerodynamics and aeroelasticity result in a stable and controllable vehicle for all possible flight conditions. The size, location, planform, and elastic characteristics of any canard control must be considered in making the control selection. Interacting effect of wing, fuselage and mass distribution can significantly impact the adequacy of the control and stability of an airplane; especially large, slender, low-load-factor airplanes.

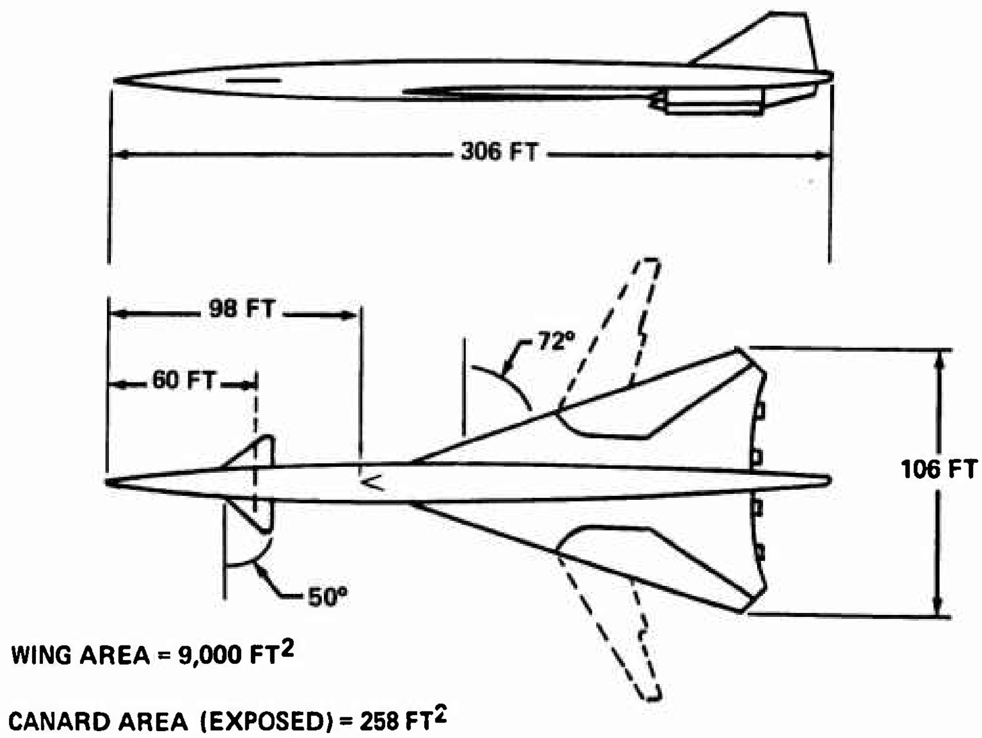

Figure 1. General Arrangement

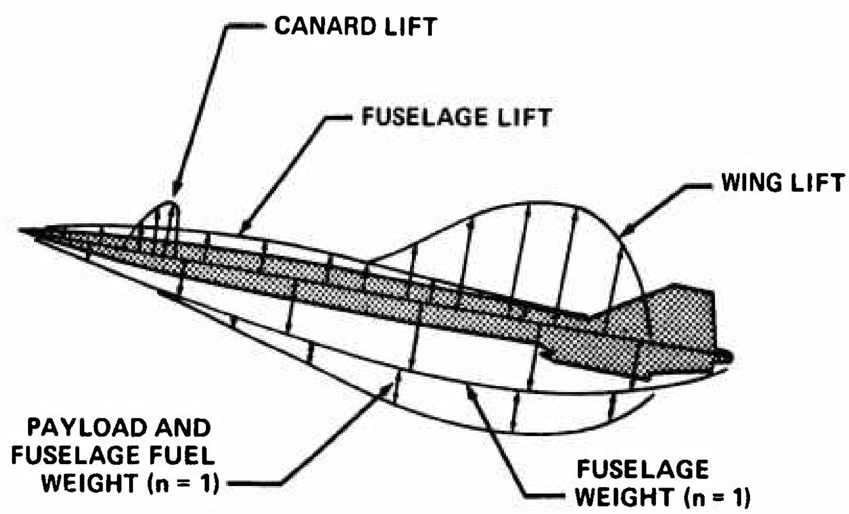

Figure 2. In-Flight Load Distributions

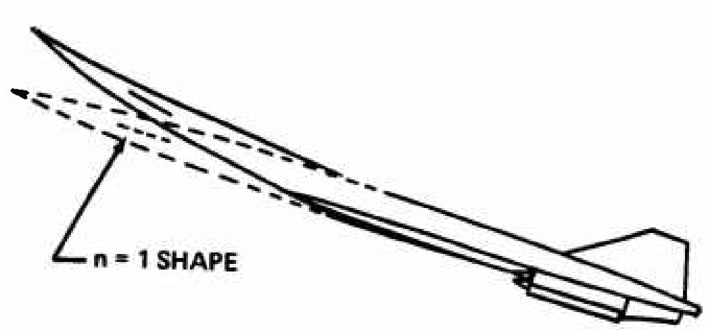

Figure 3. Light Fuselage Loading, High Normal Acceleration

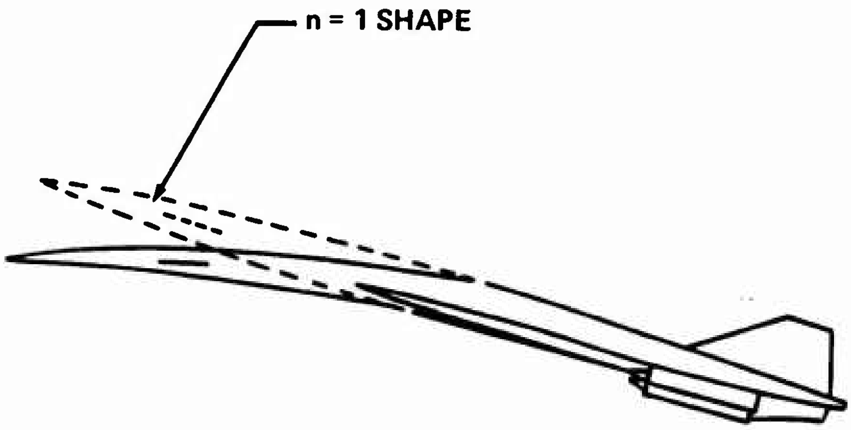

Figure 4. Heavy Fuselage Loading, High Normal Acceleration

$$ac_{\substack{\text{AIRPLANE}\\ +\text{CANARD}}} = \overbrace{ac_{\substack{\text{RIGID}\\ \text{AIRPLANE}}} + \Delta ac^*_{\substack{\text{ELASTIC}\\ \text{AIRPLANE}}} + \Delta ac_{\substack{\text{CANARD}\\ \text{RIGID}}} + \Delta ac^*_{\substack{\text{CANARD}\\ \text{ELASTIC}}}}^{\text{STATIC}} + \overbrace{\Delta MP/\hat{q}_{\text{AIRPLANE}} + \Delta MP/\hat{q}_{\text{CANARD}}}^{\text{MANEUVER}}$$

*TERMS AFFECTED BY INERTIA

Figure 5. Total Aerodynamic Center for the Airplane with Canard Controls Fixed

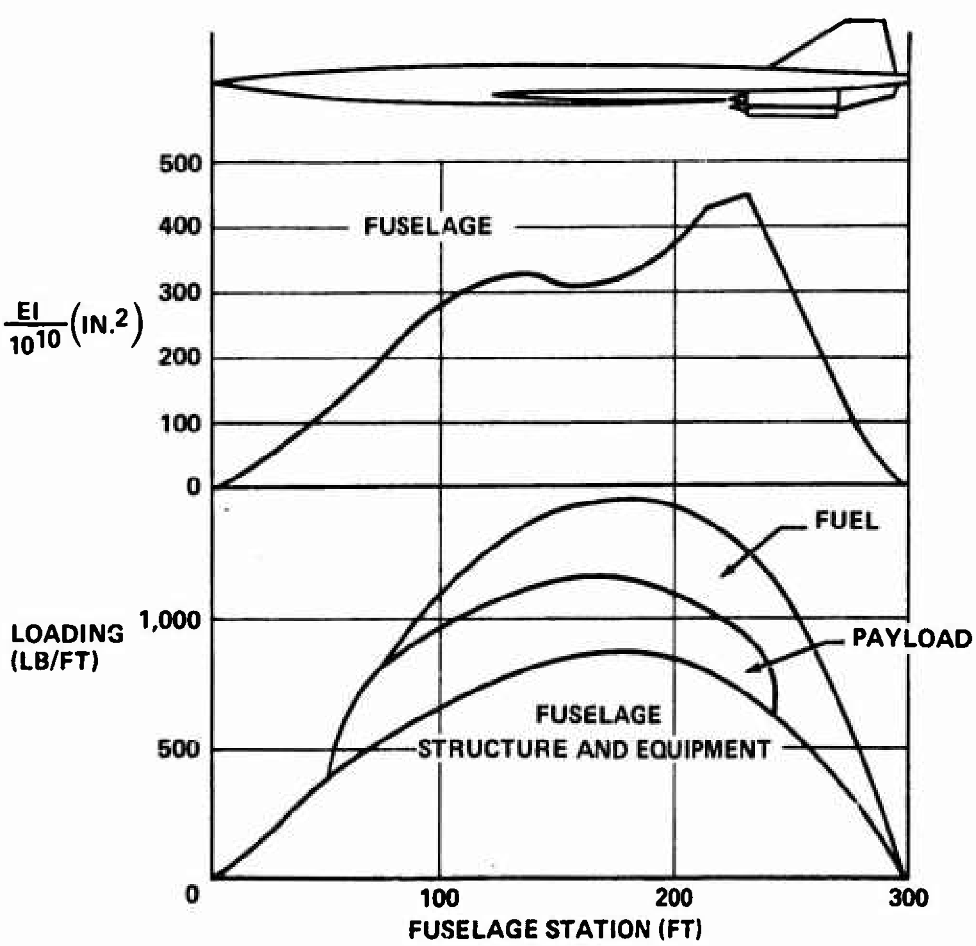

Figure 6. Fuselage Stiffness and Weight Distribution

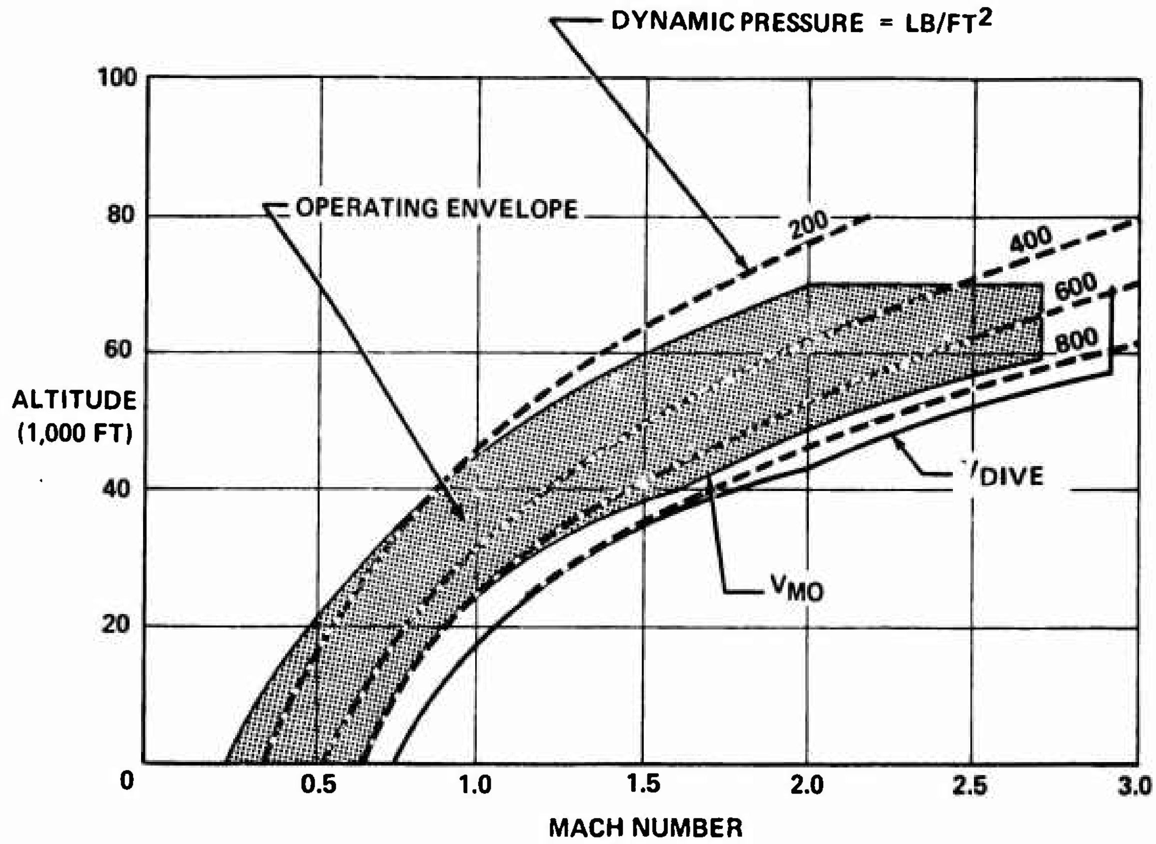

Figure 7. Flight Operating Conditions

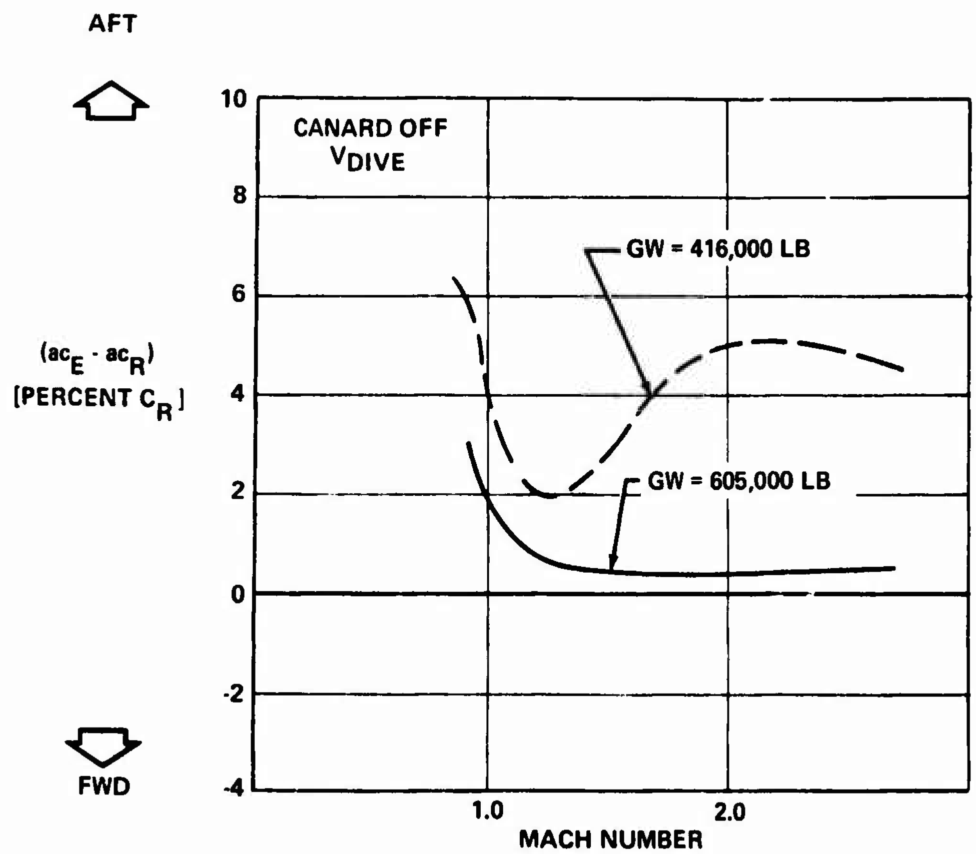

Figure 8. Aerodynamic Center Variation Due to Elasticity – Canard Off

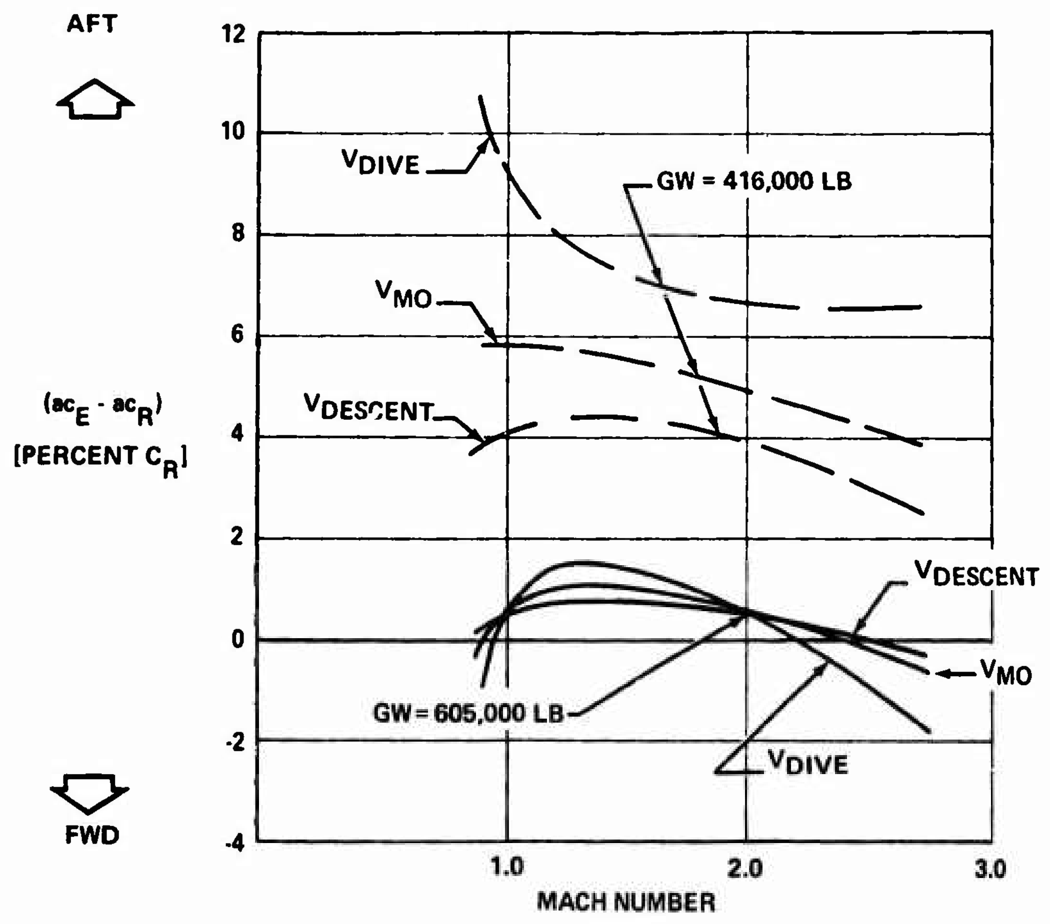

Figure 9. Aerodynamic Center Variation Due to Elasticity — Canard On

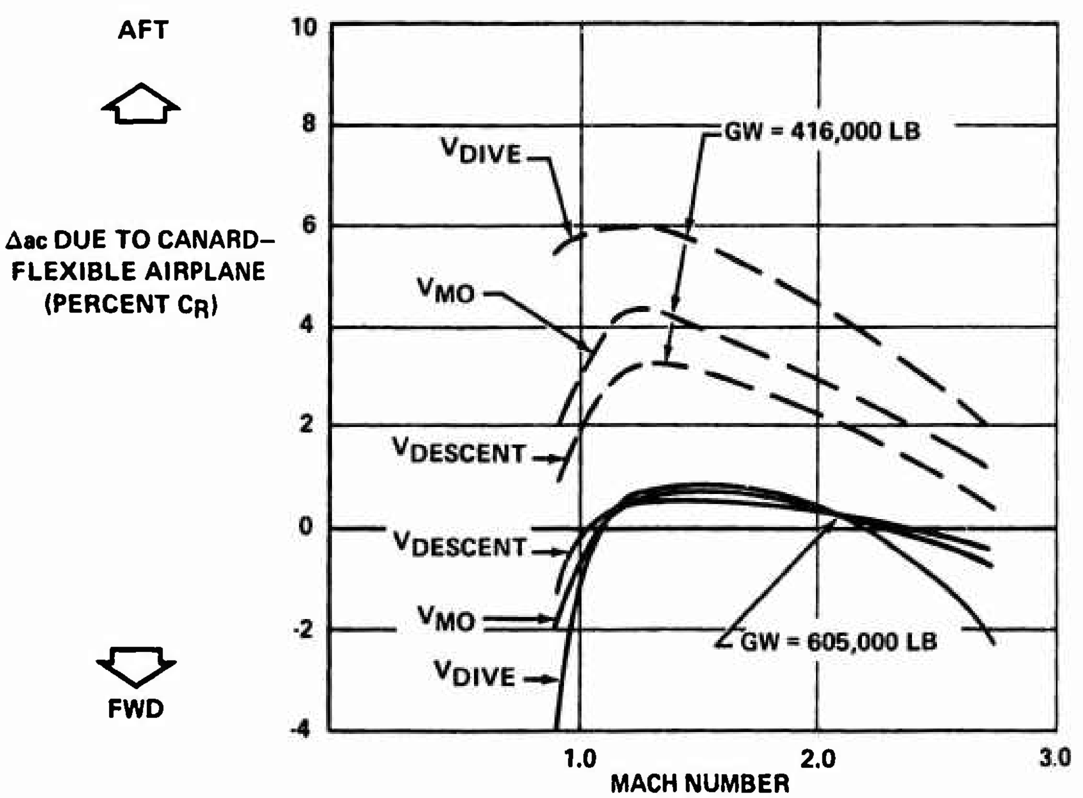

Figure 10. Aerodynamic Center Variation Due to Canard Only—Flexible Airplane

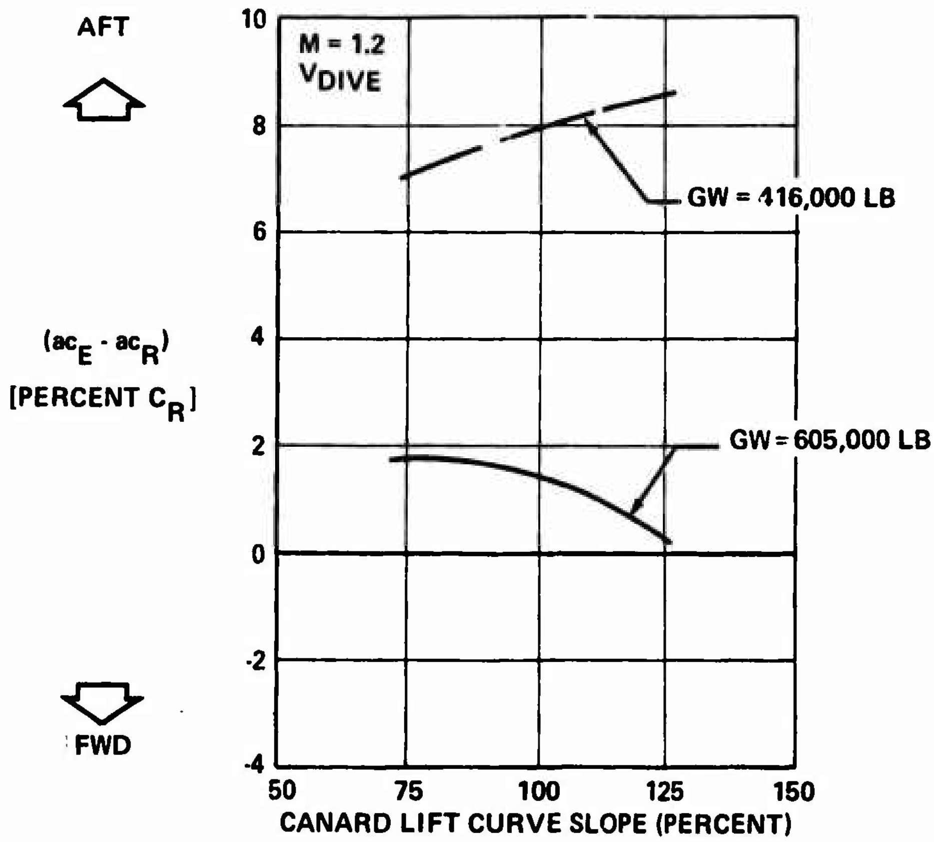

Figure 11. Effect of Canard Lift Curve Slope on Aeroelastic Aerodynamic Center Increment

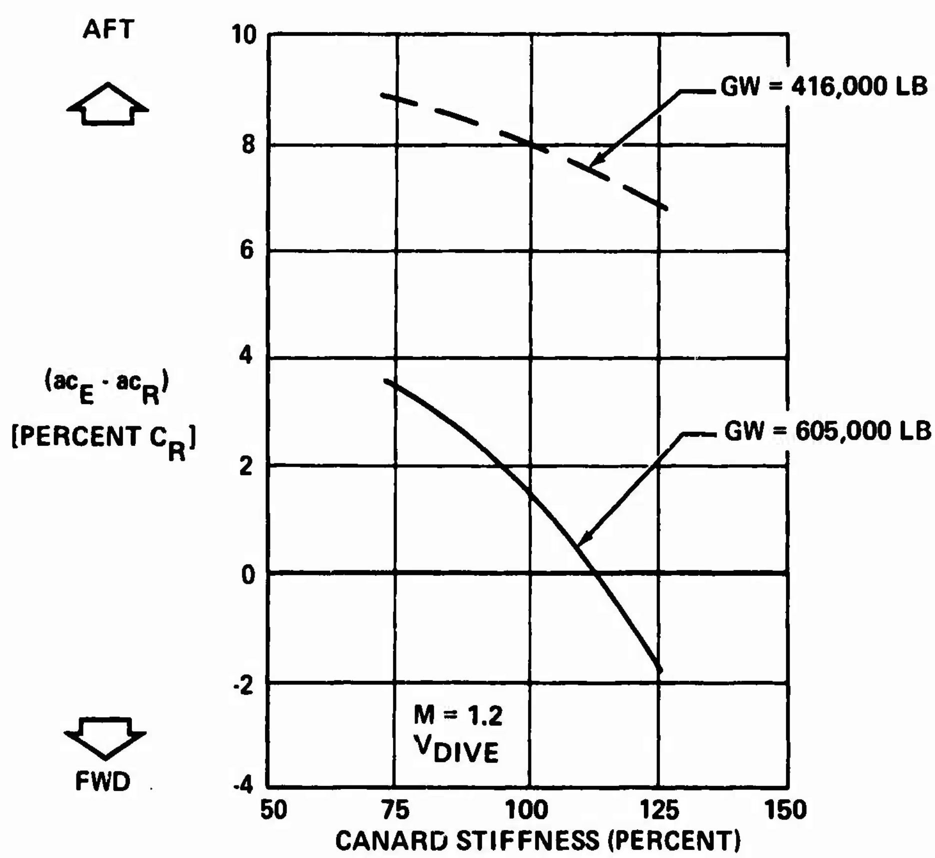

Figure 12. Effect of Canard Stiffness on Aeroelastic Aerodynamic Center Increment

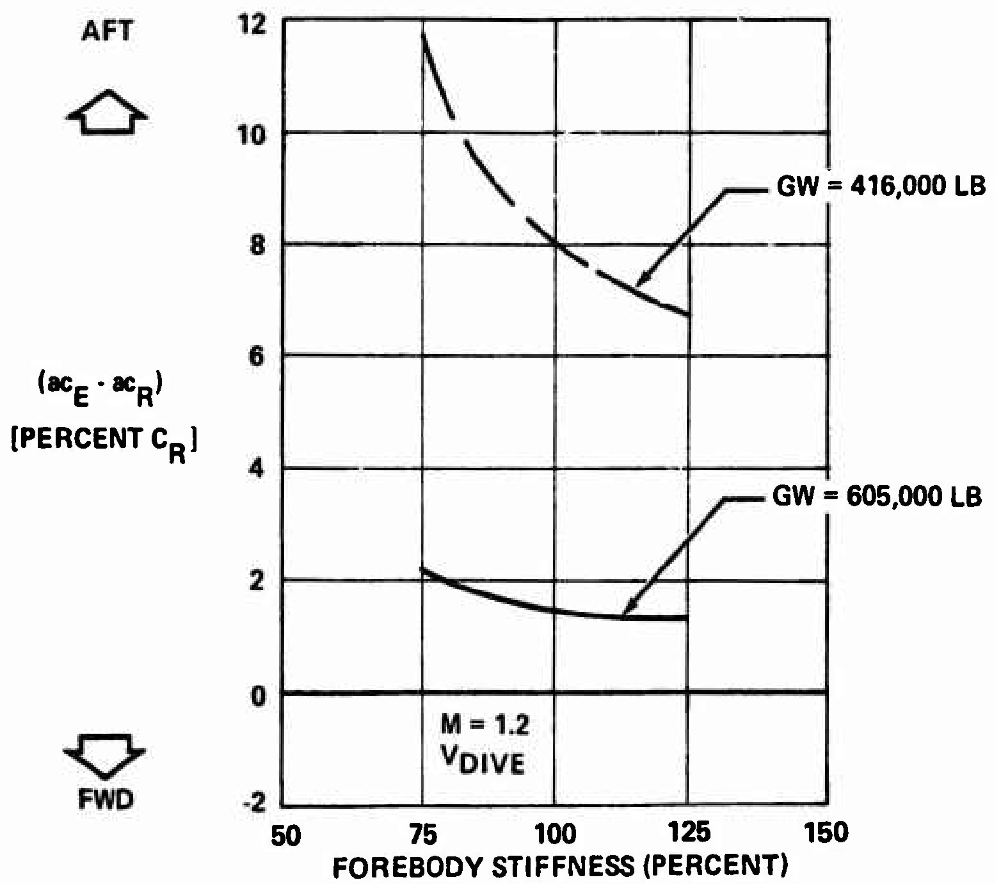

Figure 13. Effect of Forebody Stiffness on Aeroelastic Aerodynamic Center Increment

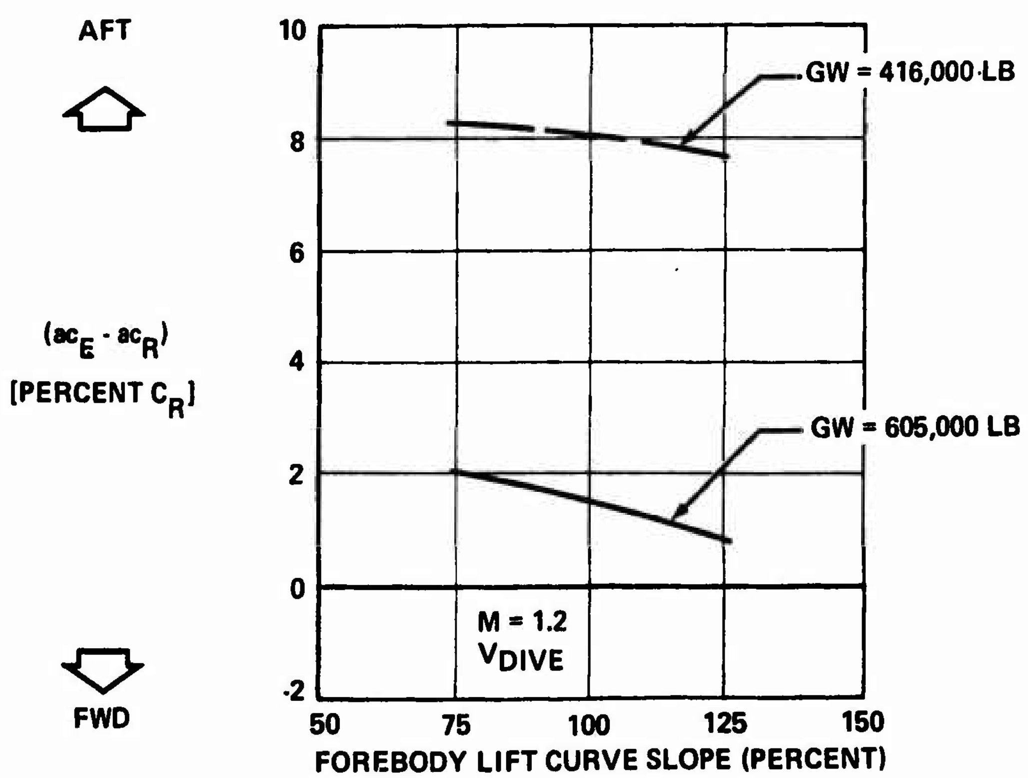

Figure 14. Effect of Forebody Lift Curve Slope on Aeroelastic Aerodynamic Center Increment

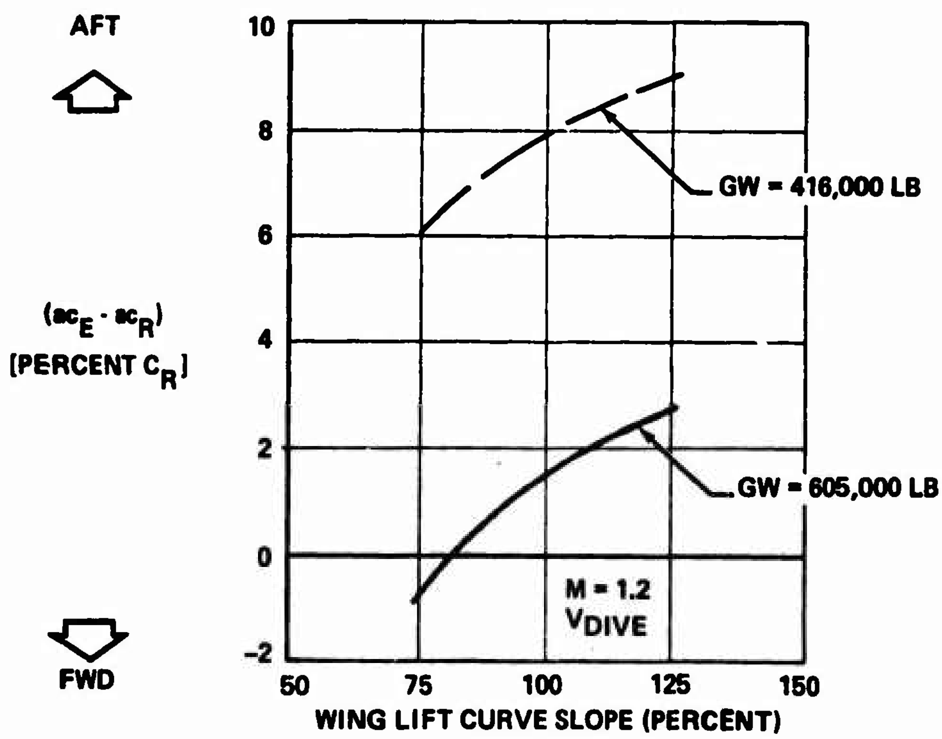

Figure 15. Effect of Wing Lift Curve Slope on Aeroelastic Aerodynamic Center Increment

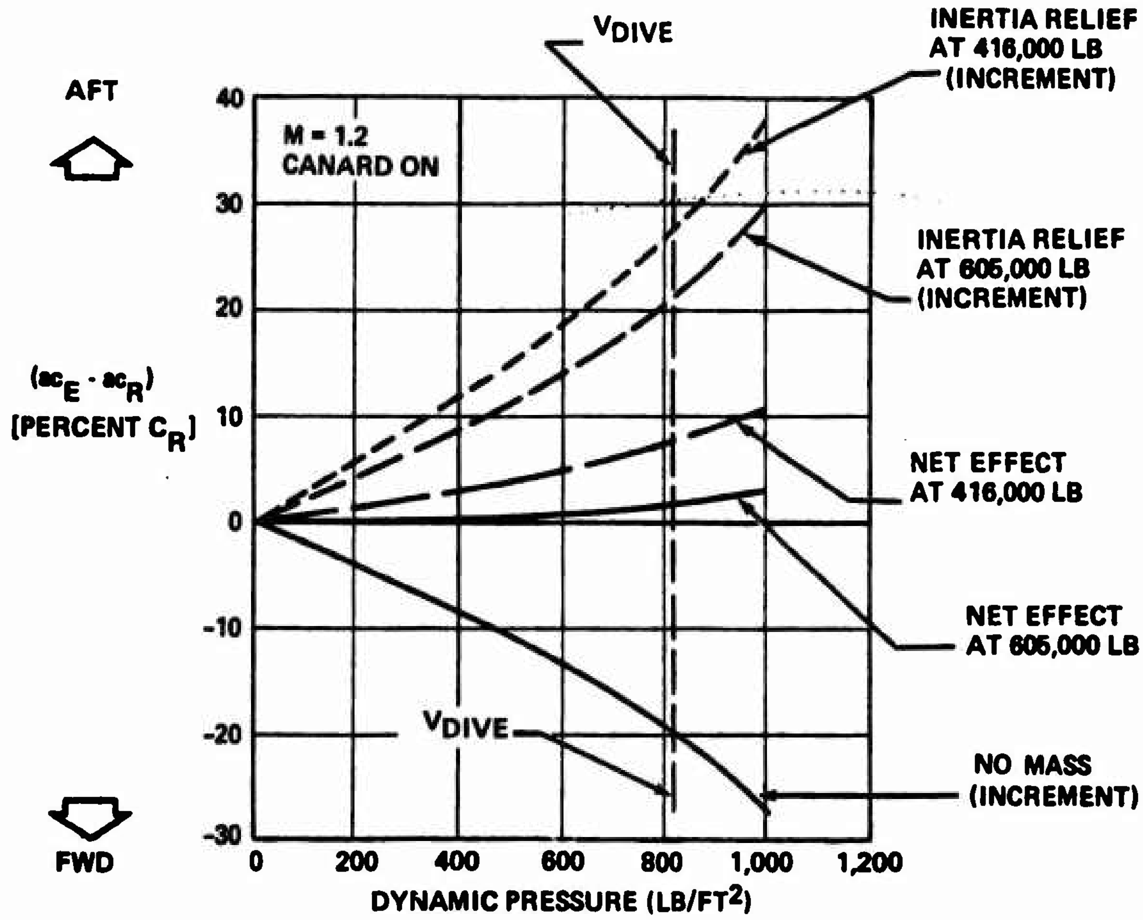

Figure 16. Effect of Dynamic Pressure on Aeroelastic Aerodynamic Center Increment

SOME FLUID-DYNAMICS CONSIDERATIONS RELATING TO THE PRELIMINARY DESIGN OF COMBAT AIRCRAFT

by

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SUMMARY

The present paper is intended primarily to highlight some of the fluid-dynamics considerations which arise in the early design stages of advanced combat aircraft. After some performance considerations of this type of aircraft, including some comparisons between fixed-wing and variable-sweep layouts, reference is made to aerodynamic problems in the design of variable-sweep aircraft which have received considerable attention during the past decade. A brief outline is given of some of the outstanding problems of wing design, wing-body interference and the estimation of wave drag. Some questions arising in the choice of engine intakes are mentioned, along with the many jet and afterbody problems, including the possible effects of jet temperature on the base drag. It is concluded that, if uncertainties and risks are to be minimised for current stringent and multi-role requirements, and if the full potentialities of the variable-sweep combat aircraft are to be exploited, then more research is needed and is well justified, especially in aerodynamics.

This paper, which had to be written at short notice, is based essentially on helpful comments provided quickly by colleagues within Aerodynamics Department, R.A.E. (Farnborough and Bedford).

SOME FLUID-DYNAMICS CONSIDERATIONS RELATING TO THE PRELIMINARY DESIGN OF COMBAT AIRCRAFT

1. Introduction

As a contribution associated primarily with the Fluid Dynamics Panel interests, the present paper is intended to highlight some of the fluid dynamics problems which arise in the early design stages of combat aircraft. These aircraft are here taken to be of tactical/fighter type. Their layout follows classical concepts with separate wings on a fuselage, and with one or two engines usually installed within the fuselage. Their operation combines a wide variety of different requirements, so that wings with variable sweep offer a promising solution.

Because such aircraft have to meet a wide range of requirements, their design cannot be biased heavily towards one predominant aerodynamic condition (e.g. the high-altitude cruise), as in the case of transport aircraft for long or medium range. In general, the combat aircraft have to operate effectively at much higher speeds and much higher/lower incidences, relative to a primary design condition, than would be the case with the transport aircraft. In this context, reference may usefully be made to the recent AGARD Specialists' Meeting(1) on transonic aerodynamics, and to an earlier International Congress on Subsonic Aeronautics(2), at which many problems were discussed which are relevant here. However, in the present state of knowledge, the designer of combat aircraft appears to be less well served than the transport designer in that he has to face even more intricate aerodynamic problems, entering into flow regimes which are even further or more frequently away from any single ideal condition. There is thus much room for additional aerodynamic research aimed at improvements in high-performance combat aircraft design.

2. Some Performance Considerations

In the classical design of swept-wing aircraft, the shape is so chosen that the simple attached and shock-free aerofoil flow is maintained ideally throughout the flight envelope, with only small excursions beyond these limits. The same aerodynamic design principles can be applied to wings with variable sweep with the result that, to a first order, variable-sweep aircraft can be operated at widely different speeds from low-subsonic to supersonic without losing range (Fig.1)(3). Thus, the low-speed characteristics and the endurance of a fixed-wing aircraft of low sweep can be combined with the supersonic capability of one with high-sweep. These basic features can be usefully exploited for military applications but, with combat aircraft, the requirements are especially demanding to ensure optimum effectiveness for radically different missions and flight regimes. For instance, extra demands are made on the manoeuvrability of the aircraft which may lead beyond the classical flow limits into regions where strong shock waves and flow separations may occur. Again, to cope with extreme demands of performance and handling requirements, there is the over-riding need to achieve an acceptable compromise between detailed design features (such as camber and twist) which may conflict with one another, even though the overall features are consistent and remarkably simple and although aero-elastic deformations can be beneficial. It is not surprising therefore that many complex and novel problems have to be tackled.

With regard to estimating performance of swept-wing aircraft, there is inevitably a great difference in complexity between the conventional long-range subsonic transport aircraft and the multi-role combat aircraft. Thus, performance studies of combat aircraft must include particularly elaborate computer programs for calculating the various missions and the effects of the large number of parameters upon these. Parametric studies, using both simplified exploratory analysis and such complex programs, have been made to determine what is technically possible and to check whether certain specified requirements are realistic. However, these studies are strictly outside the scope of the present paper and only a few results are included for the purpose of illustration, as background to the discussion of relevant aerodynamic problems.

For example, one of the main design parameters for specification is the wing loading. With conventional transport aircraft, the design wing loading is primarily determined by the required landing speed, taking into consideration of course the requirements for take-off distance and cruise, together with the high-lift devices available and the engine-thrust characteristics under various conditions. For a multi-role combat aircraft, the choice also depends on the desired manoeuvre capabilities over a wide spectrum of speed and altitude, response to gusts at high E.A.S., matching of subsonic and supersonic range, speed and rate-of-climb capabilities, etc. Fig.2 illustrates simply how, for a combat aircraft with prescribed engine size, the design wing loading at take-off first increases with aircraft all-up-weight (greater fuel-load and range) when the landing is the controlling requirement. However, the resulting increase in take-off distance eventually becomes unacceptable, so that the design wing loading must reduce steadily with further increase in all-up-weight. The ability to develop a prescribed normal acceleration (ng) during combat also provides an upper limit to the wing loading, while the gust response in low-altitude high-speed flight defines a lower limit. Moreover, in order to maintain the required high-speed performance in level flight or high rate-of-climb for interception, the resulting growth in engine size with increasing all-up-weight introduces further limiting considerations.

More generally, not all such considerations will be compatible, so that some compromise in and relaxation of initial operational demands may be necessary. Furthermore, the interplay between a large number of fundamental design parameters and such diverse performance requirements needs to be examined thoroughly, with a view not only to optimisation of cost-effectiveness, but also as regards sensitivity to possible deviations from the basic technical assumptions and mission specifications.

For our purposes, the performance capabilities of fixed-wing combat aircraft ought also to be compared against those of variable-sweep type, based on assumptions which correspond to the current state of technology. Some typical results for radius of action, and the time taken to climb to M ≈ 2 at high altitude, imply that the variable-sweep configuration is far superior (Fig.3), assuming comparable levels of short field performance are maintained. For a given weight (or cost) the fixed-wing aircraft does not provide as good radius and climb performances as the variable-sweep aircraft; alternatively, the weight of the fixed-wing aircraft to ensure the same levels of performance is much greater than for the variable-gometry configuration. This is not to argue that the same will always be true, but rather to illustrate our special interest in variable-sweep aircraft for multi-role combat aircraft operations.

3. Wing Design

In principle, the wings of variable-sweep combat aircraft can be designed according to the same concepts and methods as fixed-wing aircraft; these have already been described thoroughly by Bagley(4), and by Lock and Bridgewater(5). Many relevant matters have also been discussed at the recent AGARD meeting on transonic aerodynamics where, in particular, Haines discussed the factors affecting the basic choice of sectional pressure distributions and swept isobar patterns for economic operation at high subsonic speeds. However, very little was said there which specifically refers to combat aircraft. Here, time and space permit only a very brief outline of existing and possible design approaches.

Naturally, current practice must attempt to make the best use of available information, with intelligent application of existing theoretical and experimental techniques. Unfortunately, design details have still largely to be finalised by extensive wind-tunnel tests involving ad-hoc checks and modifications, primarily concerned with getting the overall aerodynamic forces and moments right, taking into account practical constraints on the wing geometry. Of course, some aerodynamic features which are known to be detrimental may be avoided right at the outset, such as small aspect-ratio and high-taper at low sweep angle, and pronounced kinks in the leading-edge; i.e. a 'clean wing' with well-ordered flow characteristics as well as high efficiency is very desirable. Use can be made of methods already extensively employed for transport aircraft to obtain suitable compatible sets of the basic design parameters; e.g. thickness-to-chord ratio, lift coefficient, Mach number, and sweep angle. Similarly, novel methods for designing the basic section shape can be directly applied. To obtain a good section which can cope with the extremely wide speed and altitude requirements for combat aircraft, a designer is well advised at this stage to be less ambitious and rather more conservative as regards specific optimisations, for instance in the application of rear loading.

Some indication of the severity of the problem is shown by a typical envelope of the required values for the maximum usable lift coefficients at different Mach numbers (Fig.4). The implications of stability and manoeuvre requirements on tail size have also to be borne in mind when specifying the desired wing characteristics. The estimation of the various aerodynamic boundaries obviously poses very intricate problems, especially as regards buffet onset and acceptable buffet penetration, which can involve structural and operational aspects as well as aerodynamics. Detailed experimental investigations of specific designs, such as those made by Mabey(6) are again essential, but these are difficult to perform and the relevant results may not be available in the early design phase. At that stage, the designer should also assess the advantages of partial extension/deflection of full-span leading and trailing-edge devices already desirable to obtain the high-lift coefficients required at low speeds. This can also help achieve compatibility between the basic wing design parameters.

A more sophisticated approach for use at the early design stage would be to aim to formulate comprehensive classical treatments, attempting to design rigorously for subcritical flow (with only small departures) in all flight regimes. The details of the pressure distribution over the whole surface must then be considered, taking account of the elastic properties of the wing structure. Systematic studies of this kind do not yet appear to have been completed, but they could well throw up conflicting answers, for instance with regard to the form of camber and twist required at different values of the sweep, Mach number and lift coefficient. To resolve this, variable camber and twist might have to be incorporated along with variable sweep, taking full account and advantage of aero-elastic effects, but this possibility and the engineering implications have not yet been properly assessed. However, if the designer had such comprehensive means at his disposal, he could set himself physically realistic aims and achieve realistic values even in the preliminary design phase. In this context, another basic philosophy is worth mention, where the wing is designed for classical aerofoil flow at low angles of sweep, but for flow with leading-edge vortices at high angles of sweep when the wing joins the tailplane to form a single lifting surface. Again, this possibility does not yet seem to have been explored systematically.

As an alternative design approach, we could extend the existing design methods to flows which are intended to include regions of supersonic flow with shock waves. However, the reliable treatment of such mixed flows in three dimensions requires much new knowledge, not only on inviscid flows but also on viscous interactions(7),(8), while new design criteria and aims will need to be established. Hence the development of this "mixed-flow approach" to a stage where its practical application can be seriously considered and assessed should be a subject for future research, as also should the "comprehensive classical approach" mentioned previously.

Clearly, as regards wing design and also some other aspects discussed later, our present standard of knowledge still involves serious uncertainties in the early design stage, when risks have to be taken. If these risks are to be minimised and if the full potentialities of variable-sweep combat aircraft are to be exploited, then much more research is needed and justified, especially in aerodynamics.

4. Wing-Body Interference

For the shapes under consideration, wing-body aerodynamic interference can be severe, so that the problems of wing design are aggravated by the interactions between the wings and the fuselage, while the tail unit has also to be taken into account. In view of the importance of these effects, the designer should have at his disposal methods for determining the details of the flow, such as pressure distributions over the whole surface combination, and for designing shapes which possess prescribed aerodynamic characteristics. Such methods are not yet available, except for much simplified shapes, but one can try to determine overall forces and moments.

In 1957, Pitts, Nielsen and Kaatari[9] developed a useful approach to the problem of calculating the lift and the position of the centre of pressure of wing-body-tail combinations at subsonic, transonic and supersonic speeds, using slender-body theory for the interaction terms. In the meantime, the range of such slender-body theory solutions has been extended. For example, Bartlett[10] has dealt with the wing mounted in an arbitrary way on a circular fuselage. More recently, Andrews[11] has treated wings mounted on top of fuselages of rectangular cross-sections (as in Fig.5), which approximate more closely to many combat aircraft layouts; the sides of the fuselage are assumed parallel and the bottom of the fuselage is parallel to the wing at the station considered. Convenient parameters are λ relating the breadth of the fuselage to its depth, and A^* relating the cross-sectional area to the square of the span. Then L^*, the ratio of the lift to that of a flat plate of the same span, can be found numerically. Some results are shown in Fig.6.

The horizontal line $L^* = 1$ corresponds simply to bodies with elliptic cross-sections, becoming thicker as A^* increases from left to right. The top curve corresponds to rectangular bodies in isolation and comes from the classical work of von Karman and Burgers. Andrews' results for rectangular bodies of fixed proportions, occupying a varying proportion of the wing span, run from $L^* = 1$ at $A^* = 0$ initially downwards and then upwards to terminate on the von Karman curve. The familiar curve for a wing mounted symmetrically on a circular body is also included, together with Bartlett's results for a wing on top of a circular body, (which lie very close to Andrews' curve for $\lambda = 1/2$), and Portnoy's results[12] for a semi-circular body beneath a wing.

When the overall interference effects can be as large as indicated in Fig.6, it is not surprising that local effects may be larger still. As an example, Fig.7 shows the spanwise loading over a simple 'rigid model' simulating a combat aircraft configuration with a high wing on a body of rectangular cross-section. There is a significant loss of lift near the wing root, which could be eliminated by a suitable application of camber and twist, possibly with some alleviation from aero-elastic effects. But the designer can accept such modifications only after determining what the consequences are at other flight conditions and sweep angles, and what mechanical complications are involved. This example demonstrates clearly the advances which need to be made in the aerodynamic design before the designer has a complete and reliable method at his disposal.

The foregoing discussion relates strictly to interference effects under high-speed conditions at low lift-coefficients. It must be added that reliable theoretical treatments are also needed for swept-wing body combinations at high lift-coefficients, both without and with high-lift devices extended.

5. Wave Drag

For the practical estimation of aircraft drag at supersonic speeds many contributions have to be derived, including:

> zero-lift drag, lift-dependent drag, propulsion-system drag, trim drag, excrescence and store drags.

The designer is indeed faced with a formidable task, especially if aircraft performances have to be guaranteed, because any uncertainty in drag estimation may lead to installation of extra thrust (increased aircraft weight) or ultimately necessitate extensive modifications.

Here, we shall only consider briefly the determination of wave drag at or near zero-lift, which plays a vital part in the prediction of the supersonic capability of a combat aircraft at all altitudes. This alone presents a very complex aerodynamic problem, even if it may be assumed that the contribution of the wing in its high-sweep position is small. The shapes to be dealt with are generally highly three-dimensional and not smooth, while the fluid-dynamics considerations are especially complicated, involving intake and exit flows, shock waves, strong viscous effects and interactions, and probably flow separations.

While estimates for wave drag may well be best derived from total drag measurements, the provision of the latter calls for an elaborate series of tests in which engine airflow is correctly simulated, while there is also a need to check that the skin-friction drag can be correctly estimated. For isolated slender wings, Evans(13) showed that sufficiently accurate estimates of skin-friction drag can be made by using flat-plate data on a strip-theory basis. Similarly, Winter and Smith(14) showed that, for a cambered delta wing, the total skin-friction drag was only some 5% to 10% less than that on an equivalent flat-plate, although in some areas the local skin-friction drag fell to as low as one-half the flat-plate value. Nevertheless, it has yet to be shown that the skin-friction drag can be as reliably estimated for a complex shape such as a combat aircraft. Thus, taking also into account the difficulties of correctly representing the engine flow, it is clear that a theoretical method for estimating wave drag is desirable, in order to back up the analysis of the tunnel measurements. Such theoretical estimates are in any case essential at the early stage of a design, before tunnel results are available, even if some empirical factors based on past experience have to be introduced.

Currently, the only theoretical methods applicable make use of linear theory, with additional assumptions leading to the supersonic area rule and transfer rule. These are formally equivalent, but the transfer rule gives scope for including empirical values of wing drag, although it is debatable whether this necessarily leads to greater overall accuracy of prediction. Both rules involve the evaluation of von Karman's double-integral, for which the methods of Eminton or Cahn and Olstad can be employed. Unfortunately, such applications are unsatisfactory in principle, because the physical flows involved do not normally satisfy the assumptions of the linearised theory. Local disturbances are large, particularly at the aircraft nose, canopy, and wing leading-edge, so that not only does the theory incorrectly estimate the effect of surface slopes but also does not properly account for the propagation of the disturbances.

Furthermore, apart from such fundamental fluid-dynamics objections, the methods have been shown to be somewhat unsatisfactory in practice, for example the data presented by Harris(15) showed that, for a simple body of revolution with a fineness-ratio typical of combat aircraft, errors as high as 20% could occur in the drag estimates. Surprisingly, his estimates for complete aircraft shapes are rather better, suggesting the presence of a compensating error in these particularly simple examples. In addition, difficulties arise in specifying the area distributions to be used in these methods, particularly where discontinuities in shape occur, such as at the canopy and wing leading-edge, since these discontinuities will be smoothed out to some extent by the boundary-layer. Another difficult region for interpretation is at the propulsive nozzle, where the presence of the expanding jet may induce separations ahead of the rear end of the aircraft structure.

Some of the objections could be removed by the development of more exact solutions for the inviscid part of the flow. For example, an exact solution of the linear theory equations has been derived by Woodward(16), and makes use of singularities on the surface; however, the fundamental objections to linearised theory can still apply. The further development of practical treatments based on the method of characteristics would appear feasible for non-smooth non-symmetric shapes, though possibly expensive and time-consuming. Moreover, such improvements would need to be supplemented by improved methods for dealing with three-dimensional turbulent boundary-layers including flow separations and viscous interactions.

6. Air Intakes

Most combat aircraft designs feature single or twin engines mounted in the fuselage and fed by air from intakes on the sides of the fuselage. In general, the engine flow is relatively large and significantly affects the flow past the whole aircraft. In particular, the intake flow may interact with the flow past the fuselage and the wing, and the jet may interact with the flow past the afterbody. The intake must operate over an extraordinarily wide range of flow conditions, and the basic design problem centres mainly on the compromise to be struck between these, especially between the demands of efficient high-subsonic cruise and high-altitude supersonic interception. Thus, major problems can arise from the need for:-

a) Reconciliation of the conflicting requirements for low spillage drag at fairly high rates of spill in low-altitude flight at high-subsonic speeds, with those for low external wave drag and low rates of spill (together with high internal pressure recovery) in high-altitude supersonic flight;

b) Provision of an internal flow with high pressure recovery and uniform pressure distribution at the engine face under all flight conditions, including the extreme conditions of attitude that a highly manoeuvrable aircraft will attain.

At supersonic speeds, at least, an attempt can be made to design the compression surfaces in such a way that a desired system of shock waves is realised. Design methods are available for relatively simple shapes, such as two-dimensional and axisymmetric intakes. But off-design conditions and boundary-layer diverters and bleeds pose so many additional problems that extensive experimental investigations are indispensable. To execute these at a reasonably early point in the design cycle requires powerful and versatile experimental rigs.

Hence, to enable pressure recovery and flow distribution to be investigated in detail, a 'Generalised rig for testing intakes (GERTI)' has been developed by Goldsmith[17], at R.A.E. Extensive studies have already been made with this rig on three relevant types of intakes:- a rectangular intake with a variable second-wedge compression surface; a similar, but half-axisymmetric design with a variable second-cone angle; and another half-axisymmetric intake with a translating single-cone compression surface. Under some conditions of intake design and test Mach number, simple theoretical predictions of maximum capture flow and pressure recovery are possible, and reasonably good agreement between estimates and measured results can be achieved, as illustrated by Fig.8.

As regards drag contributions, major areas of inadequate data currently being studies at R.A.E. include:-

a) drag of fuselage boundary-layer diverters or bleeds, at subsonic and supersonic speeds;

b) influence of cowl shape and compression-surface geometry on spillage drag, at both subsonic and supersonic speeds;

c) definition of cowl shapes which effectively compromise between low spill drag at high-subsonic speeds and low wave drag at supersonic speeds.

7. Afterbody and Jet Problems

The aerodynamic problems associated with the back end of a combat aircraft are also especially complex and, apart from a few limited theoretical treatments, their solution again requires experimental studies using specialised and versatile rigs. A major design difficulty is to reconcile the high base drag associated with a large base area, presented when the nozzle is in the unreheated condition at high-subsonic speeds, with the requirement for a large nozzle exit area (associated with reheat) and possibly a convergent-divergent nozzle for efficient propulsion at supersonic speeds. Ideal aerodynamic solutions are well known, but they involve considerable variations in the geometry, so that mechanical complications and weight usually preclude their use in practice. The problem is even more severe if there are two nozzles which have to emerge from a fuselage with cross sections determined by other considerations. Unless the nozzle exits are well downstream of the fin and tailplane, possible longitudinal and lateral stability changes associated with various efflux interference effects also have to be determined, taking into account the operation of nozzles over a wide range of pressures, possible changes of nozzle geometry, and engine-out conditions.

More extensive discussions of such problem areas can be found elsewhere[17]. Here, we shall illustrate only one particular aspect, namely the effect of jet temperature on the base drag, by reference to Reid's tests[18] on a single axisymmetric nozzle in a cylindrical afterbody. The results in Fig.9 imply that the jet temperature has a significant effect on base drag at high subsonic speeds ($M_\infty \approx 0.9$), particularly at low jet pressure-ratios ($P_j/p_\infty \approx 2$) with the convergent nozzle ($M_j = 1$), when wind-tunnel measurements with unheated jets can give unrealistically high base drags.

7. Concluding Remarks

This brief paper has not attempted to review the status of aerodynamic technology for combat aircraft design, but has discussed some major fluid-dynamics problems which have been highlighted by experience in recent applications of such technology. The theoretical and experimental techniques already available can provide, with intelligent and critical application, a useful basis for preliminary design studies. However, to meet the increasingly stringent performance requirements and multi-role demands, yet still minimise aircraft development delays and costs, further improvements in methods for aerodynamic design and in theoretical/empirical frameworks for aerodynamic predictions are essential. Fortunately, such improvements are in fact possible, though their achievement will necessitate substantial research investigations involving precise wind-tunnel and flight experiments as well as theoretical advances. Moreover, current development work on specific aircraft projects can also be profitably directed to this end, at the same time as attempting to achieve the specified aircraft full-scale behaviour with greater certainty, by pursuing relevant investigations with extreme care, choosing model test changes systematically from an 'aimed research technology' aspect rather than merely as hopeful 'ad hoc' fixes.

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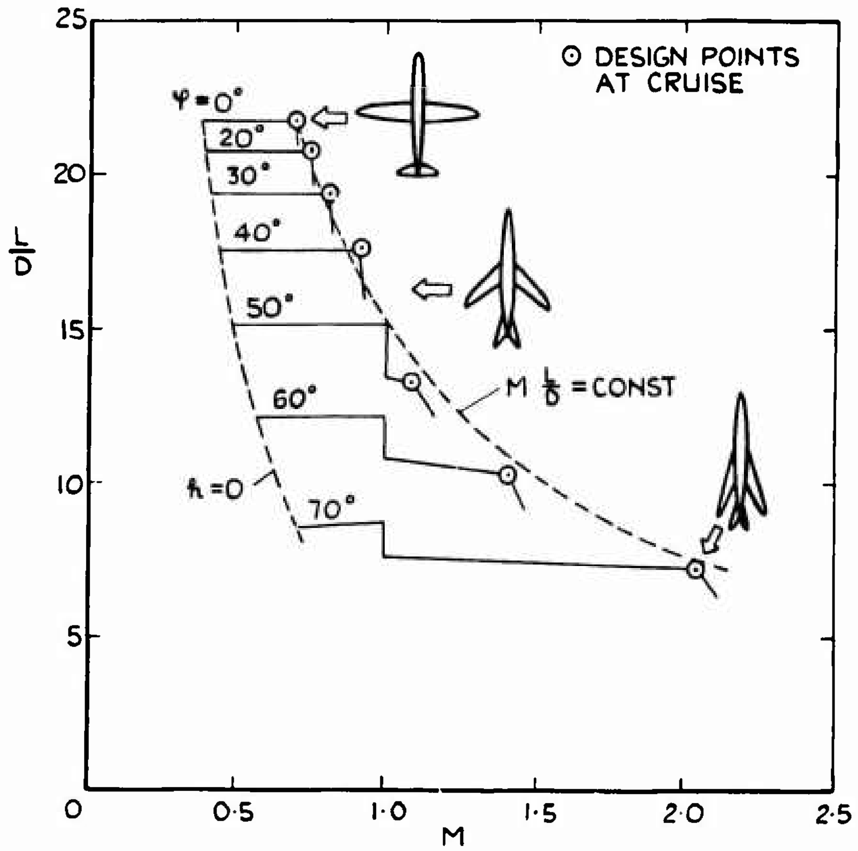


Fig.1 Lift-to-drag ratios for a family of swept-winged aircraft

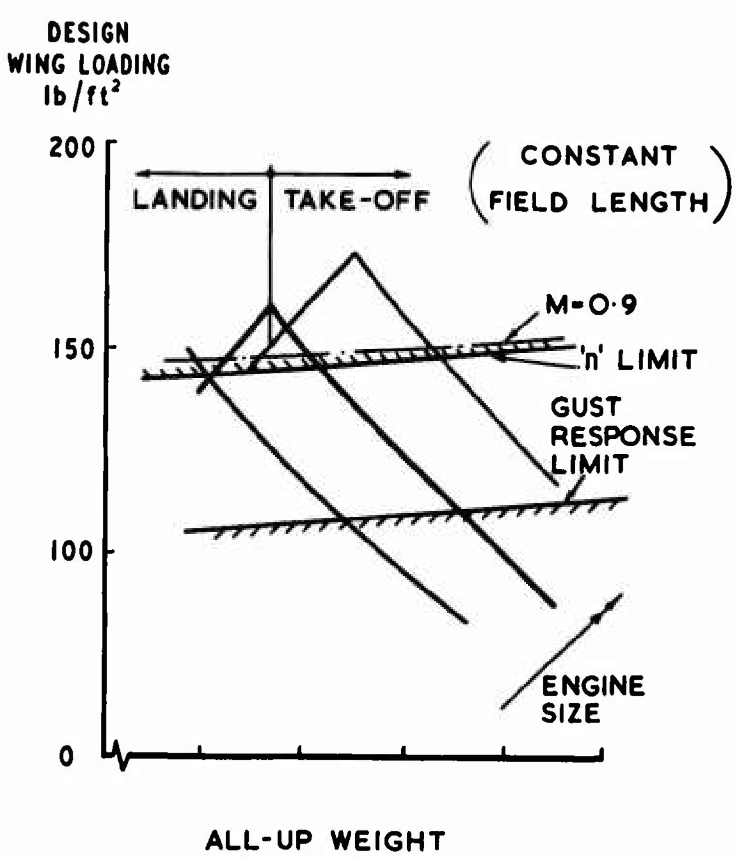


Fig.2 Effect of all-up weight on wing loading

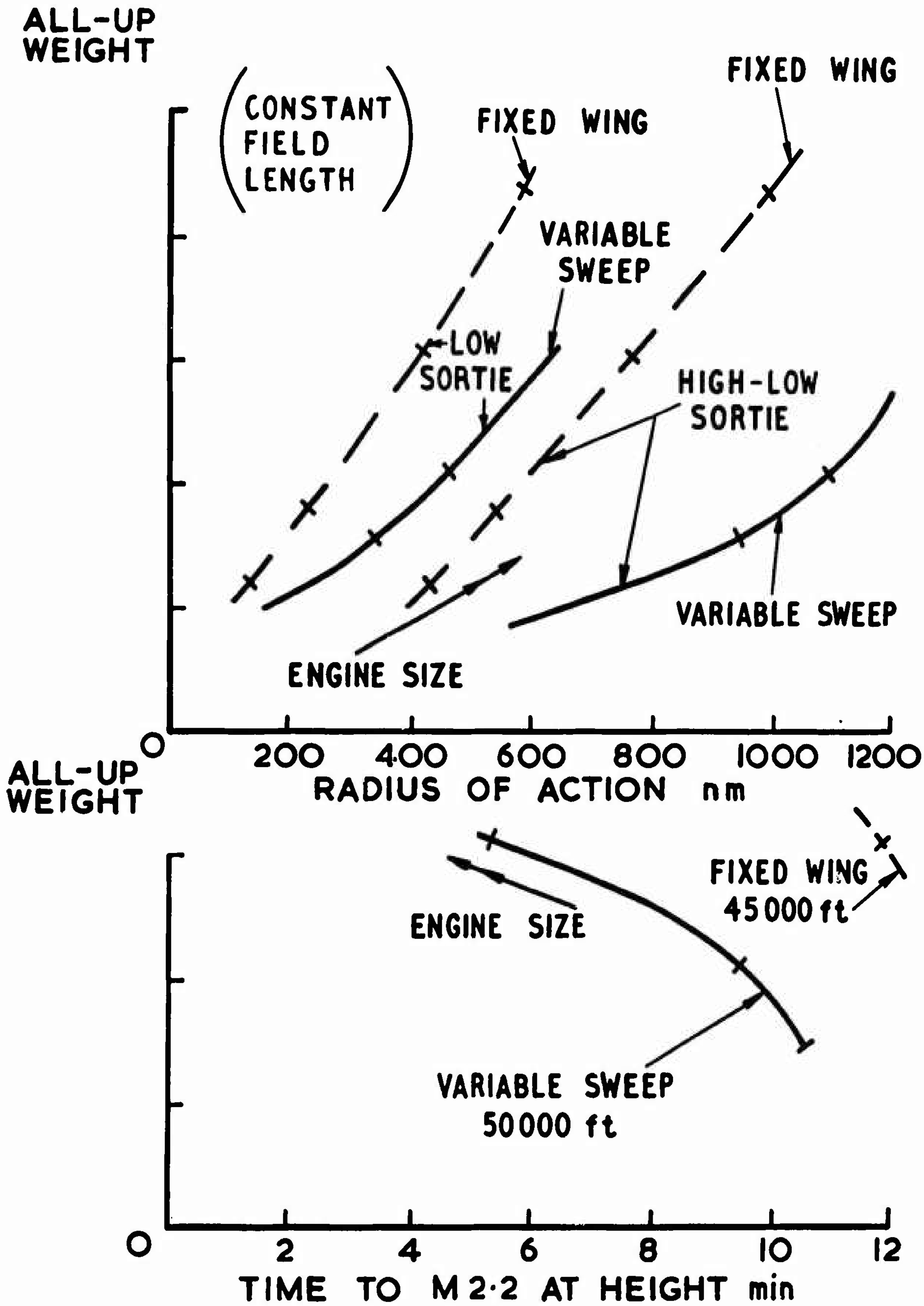


Fig.3 Weight and performance comparison variable sweep and fixed wing

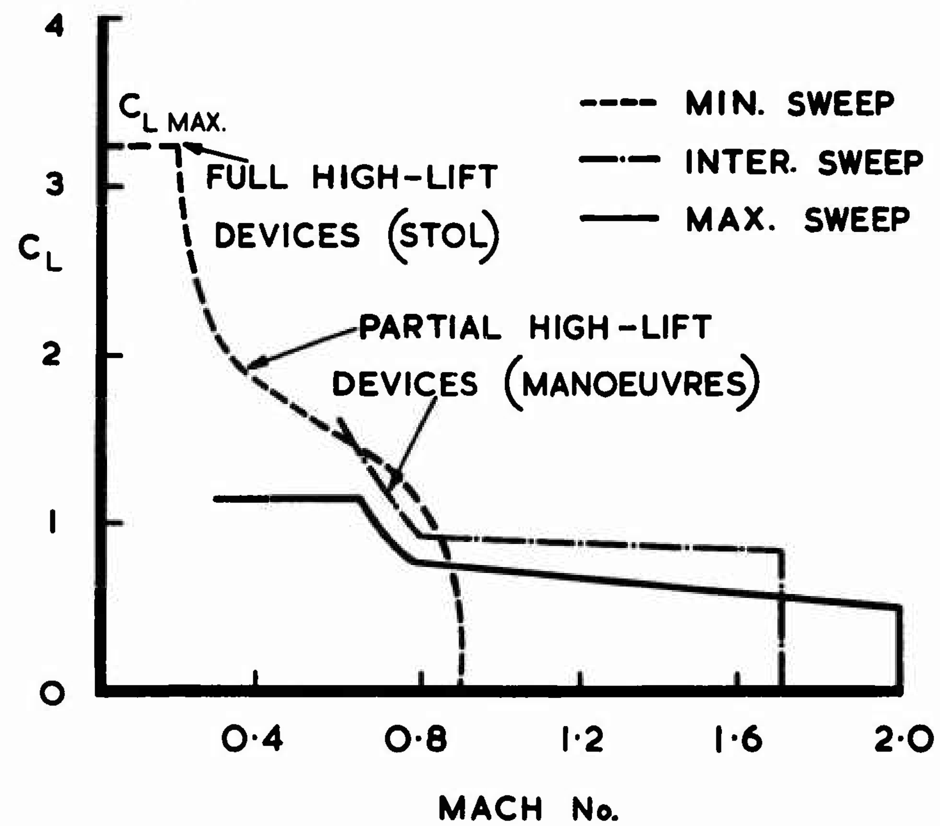
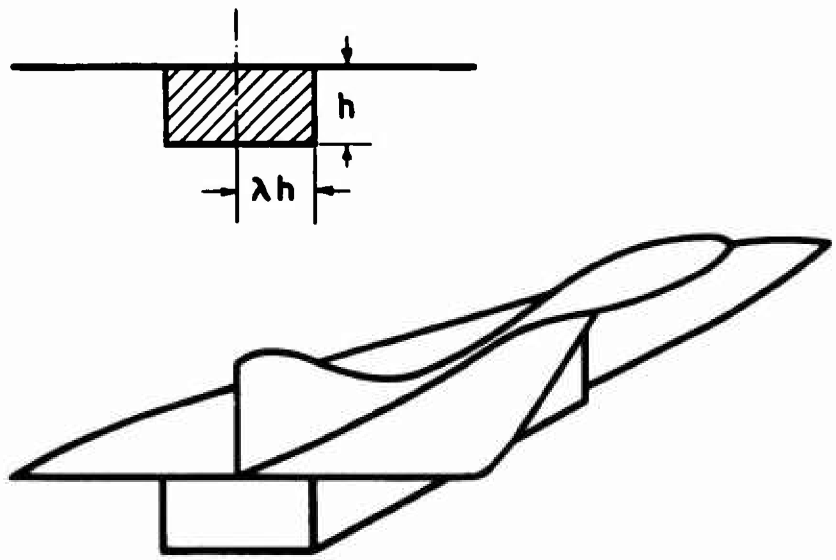


Fig.4 Possible C_L demands for a variable-sweep combat aircraft



$$A^* = \frac{2 \times \text{AREA OF CROSS-SECTION}}{(\text{OVERALL SPAN})^2}$$

$$L^* = \frac{\text{LIFT OF WING-FUSELAGE COMBINATION}}{\text{LIFT OF FLAT PLATE OF SAME OVERALL SPAN}}$$

Fig.5 Typical wing-body combination

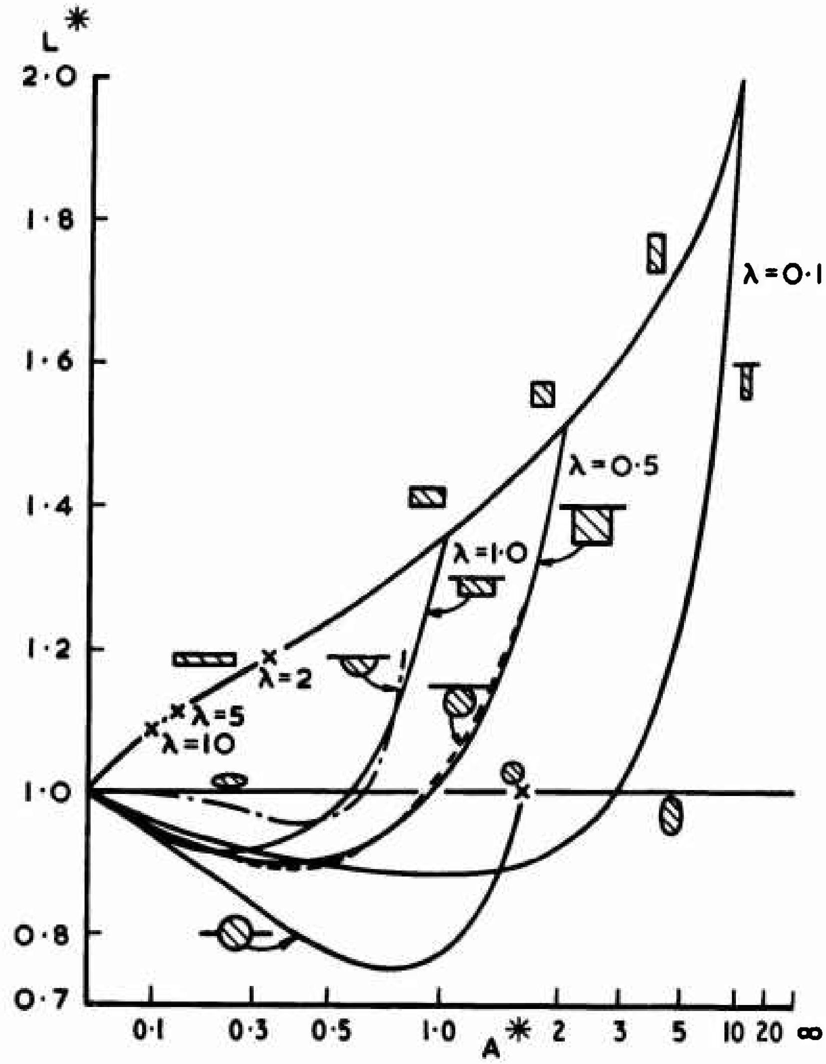


Fig.6 Lift of wing-body combinations

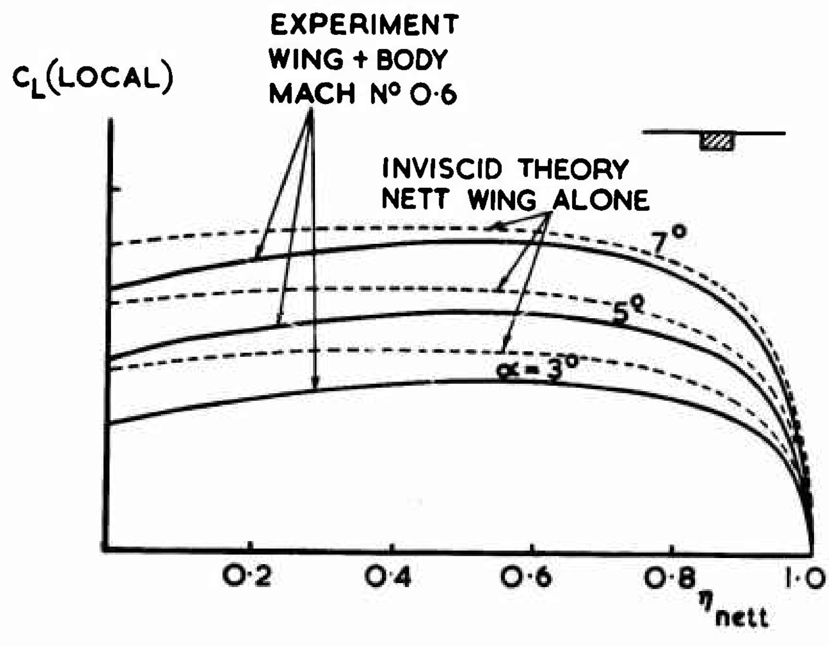


Fig.7 Lift distribution on a high wing and rectangular body combination

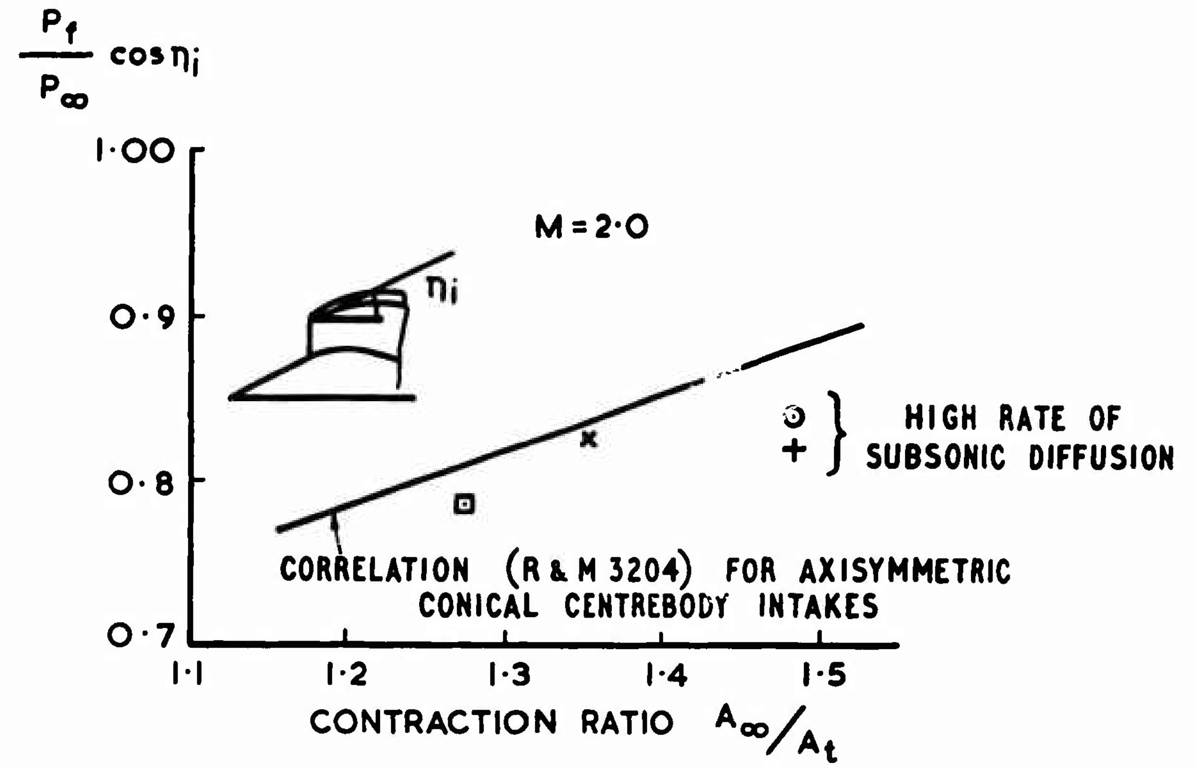


Fig.8a Pressure recovery for half conical centrebody intake

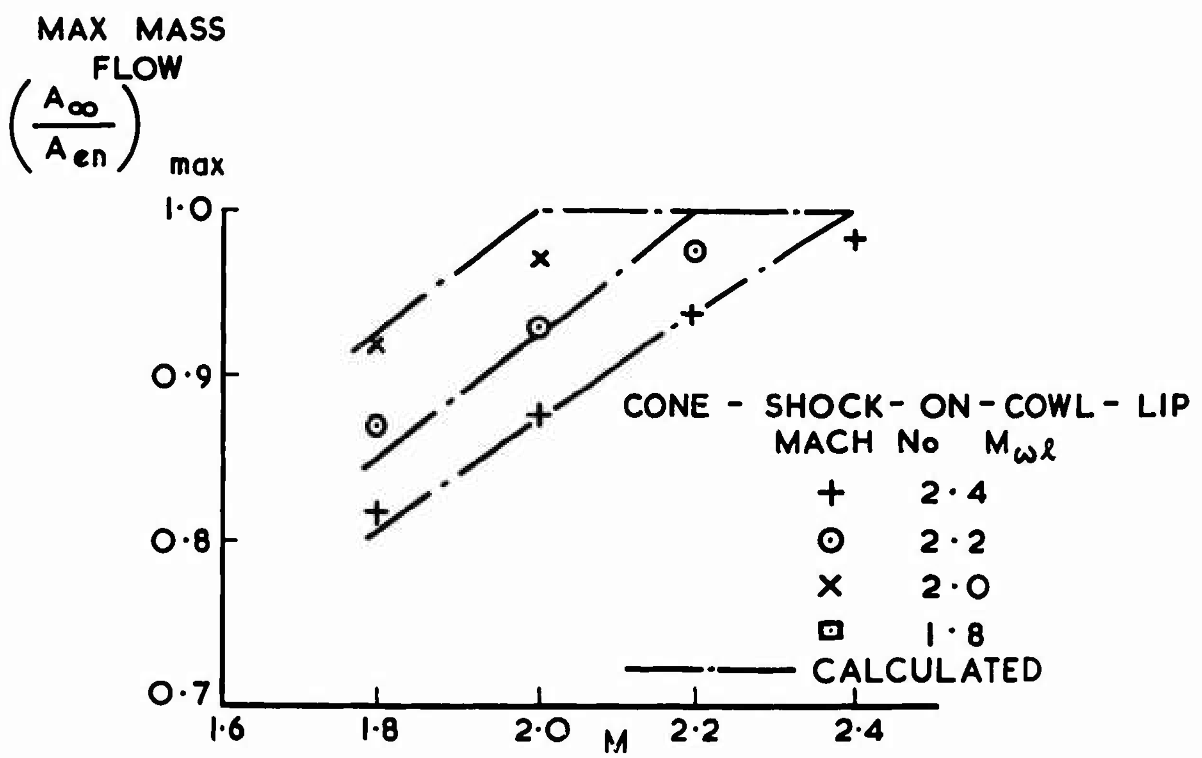


Fig.8b Maximum mass flow for half conical centrebody intake

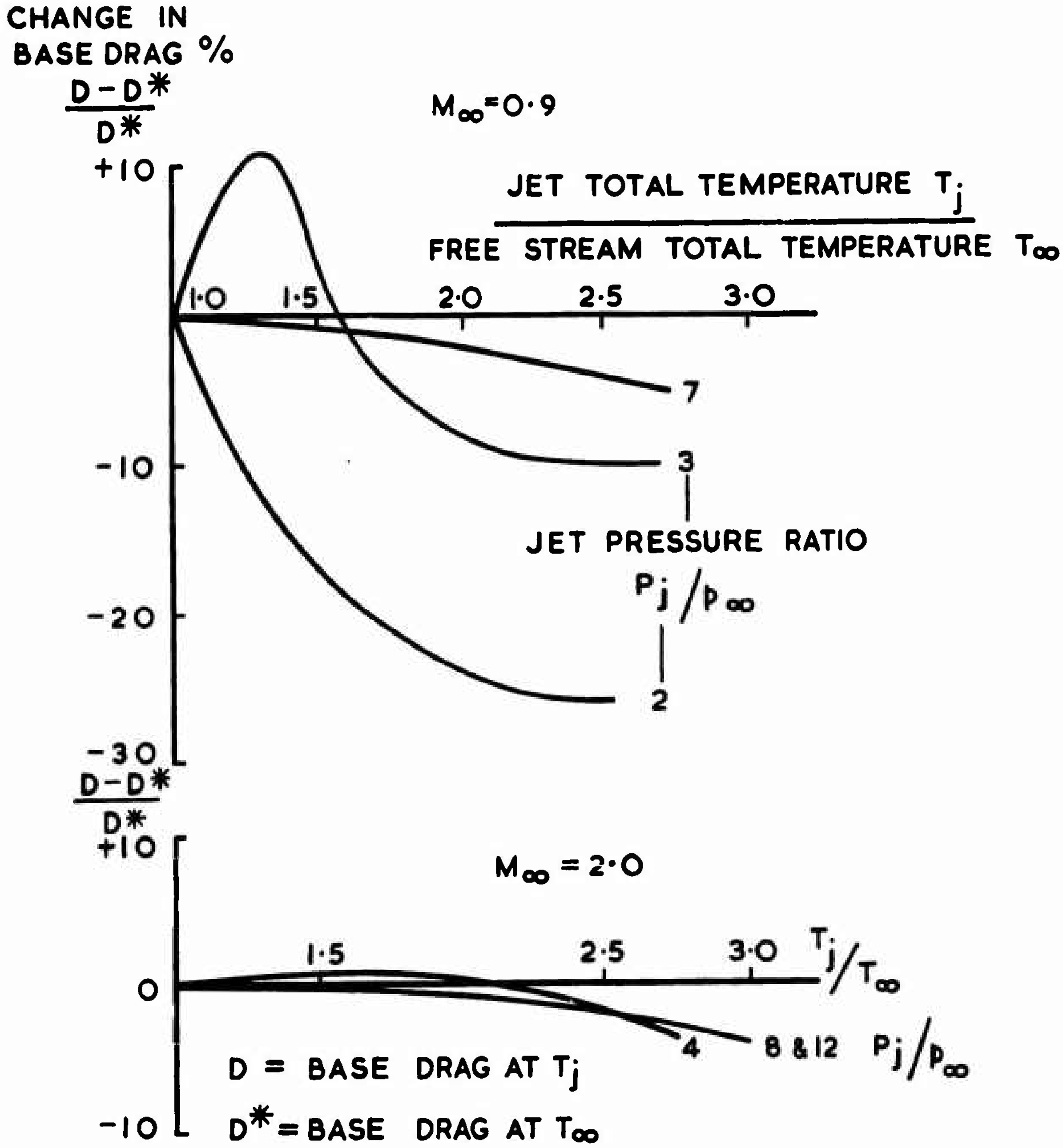


Fig.9 Effect of jet temperature on base drag (M_j = 1.0)

FUTURE ADVANCES IN THE AERODYNAMICS OF MILITARY STRIKE AIRCRAFT

by

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SUMMARY

The primary mission requirements of an air superiority fighter are reviewed and the factors which affect performance and maneuverability are discussed. The aerodynamic features which have a strong influence on fighter capability are indicated. In the last 5 years the rapid development of numerical solution techniques, using the digital computer, has revolutionized aerodynamic design methods. Future generations of fighter aircraft will have improved aerodynamic design features which will reduce the drag due to lift and provide improved stability at high angles of attack in the transonic region.

The current trend toward configurations with minimum basic aerodynamic stability and extensive stability augmentation is discussed in relation to the handling qualities requirements in the new USAF Flying Qualities Specification, MIL-F-8785B. The need for improved aerodynamic stability is emphasized and some of the current flight problems of supersonic fighter aircraft are described. It is shown that stability augmentation can cause adverse effects in some flight regimes. The analyses and test programs that are essential before an aerodynamic design is committed to production are summarized.

NOTATION

Ax	acceleration along flight path (ft/sec/sec)
Az	acceleration normal to flight path (ft/sec/sec)
g	acceleration due to gravity at sea level
T	installed thrust (lb)
D	total aircraft drag (lb)
W	gross weight (lb)
V	speed along flight path (ft/sec)
R	radius of turn (ft)
R/C	rate of climb (ft/sec)
$\dot{\phi}_T$	rate of turn
q	dynamic pressure (lb/sq ft)
C_L	lift coefficient
$W/_S$	wing loading (lb/sq)
γ	flight path angle
P_s	specific excess power (FT/SEC)

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1. INTRODUCTION

The history of fighter aircraft is generally dated from 1 April 1915, the date Roland Garras in his Moraine-Saulnier with deflector plates on the propeller blades and a machine gun on the cowling shooting through the propeller disk entered combat. The life of this configuration, which introduced to the air forces of the world the tactics of fighter warfare, was short. It however, started a crash program by all participating forces to design increasingly superior fighter aircraft. These World War 1 machines established basic guide lines of speed and maneuverability which have remained the prime requisite of a fighter aircraft for these past fifty-four years.

2. PERFORMANCE AND AERODYNAMIC DESIGN

What are the really important performance requirements of both interceptor and air combat superiority fighters? First, both must have excellent longitudinal acceleration capabilities within their flight envelope. Second, mission range and combat time must be sufficient to accomplish the area defense mission. Third, the aircraft must have good handling qualities throughout the mission envelope and very good to excellent qualities in the combat zone. Fourth, both must be able to effectively employ their ordnance. Within these general performance requirements, many important design properties such as wing loading and thrust loading will optimize out at different values for the two types of aircraft. Still other characteristics such as wing and fuselage aerodynamic design may have many similar characteristics. In level flight the available longitudinal acceleration, A_x, is given by:

$$A_x = g\left(\frac{T - D}{W}\right)$$

In the case of the interceptor, time to minimize enemy penetration is critical. To minimize the time needed to cover a given distance the aircraft must be flown in a manner which maximizes A_x to achieve its maximum speed quickly. It must also climb rapidly. Here again, the available A_x is important since:

$$\gamma \cong \sin^{-1} \frac{T - D}{W} \cong \sin^{-1} \frac{A_x}{g}$$

$$R/C \cong \left(\frac{T - D}{W}\right) V \cong \left(\frac{A_x}{g}\right) \cdot V$$

It is obvious that an interceptor will spend much of its time at low lift coefficients where induced drag and trim drag are a minimum. Of course, any technique which will reduce the form drag and supersonic wave drag will be important to this type of aircraft. The air superiority fighter depends primarily upon its maneuvering capability for combat effectiveness, assuming other important characteristics such as ordnance effectiveness, handling qualities and structural design are not unduly compromised. Maneuvering capability is defined as the ability to achieve a high rate of turn and a small radius of turn. The rate of turn is given by:

$$\dot{\theta}_T = \frac{\sqrt{A_z^2 - g^2}}{V} \quad \text{(RAD./SEC)}$$

Thus, for a given combat speed, A_z must be maximized to achieve best maneuverability. In terms of wing loading and lift coefficient:

$$\frac{A_z}{g} = \frac{L}{W} = q\left(\frac{C_L}{W/S}\right)$$

In the process of reaching high lift coefficients induced drag and trim drag quickly become limiting factors for a given thrust unless the aircraft is placed in a dive which, of course, is not an acceptable solution. Wing loading can be minimized, but this penalizes range and payload so trade-offs are necessary. Thus, high thrust loadings combined with a high ratio of $C_L/W/S$ are important for the air superiority fighter. In addition, operational experience indicates that the ability to quickly accelerate from subsonic cruise speeds to supersonic speeds of the order of $M = 1.5$ is quite important to achieve a favorable energy level in the initial stages of combat. Here again, thrust

to weight ratio is the dominant parameter. In view of these considerations it is apparent that thrust to weight ratios must be relatively high and wing loadings should be moderate to achieve a good fighter. Within the United States a method for comparing different fighter aircraft on the basis of their "energy maneuverability" at a specific speed, altitude and load factor has been used quite extensively. Energy maneuverability is simply the instantaneous rate of climb available at a specified condition. It is usually defined as, P_s "specific excess power".

$$P_s = \left(\frac{T - D}{W}\right) V = \left(\frac{A_x}{g}\right) \cdot V \qquad \text{(Ft/Sec)}$$

Contours of constant P_s, such as shown in Fig. 1, are useful in illustrating flight regions of superior combat potential levels. This figure shows the energy maneuverability levels for a typical fighter at 3 g load factor. The line for P_s = zero represents the upper limit at which the aircraft can fly at the specified load factor and maintain constant altitude, constant speed flight. Positive values of P_s display an unused potential to modify the flight path either by climbing, accelerating or increasing the load factor. Conversely, negative values represent application of a loading in excess of the power capabilities of the aircraft; the aircraft will have to decelerate or descend to maintain flight at the applied load factor. It is apparent that when two competing designs are compared at the same load factors and same speed and altitude, the aircraft with the greater value of P_s may have a decided combat advantage. It is also rather apparent that this procedure can be used as a design tool by making this type of comparison for the various parameters such as wing loading and thrust loading.

The above system of evaluating designs while rapid in required computer time and simple to apply does not permit a detailed study of several features of the proposed design. For one thing, it does not show what amount of the lift potential is usable in steady flight or whether tail power (control) or thrust is limiting the maneuver potential. However, using the $P_s = 0$ curves for the various load factors, the areas of influence of various limits can be identified. Contours of $P_s = 0$ for several load factors are shown on Fig. 2 as solid lines. Superimposed are a series of dashed lines which represent maximum possible aerodynamic load factors. Line A on this curve defines the point above which at any particular load factor excess thrust deficiency ($P_s = 0$) becomes the limiting factor.

Above this line the aircraft can temporarily attain higher g's through increasing angle of attack up to C_{LMax}, than can be supported by excess thrust. The region between $P_s = 0$ and the C_{LMax} load factor represents a negative performance potential where loss of speed and or altitude will occur during a maneuver. The area above Mach 1.2 where these aerodynamic load factors level out at constant altitude are limited by tail power on this aircraft. The ability of the aircraft to perform constant altitude, constant speed turns is essential if the aircraft is to be capable of sustained combat with another aircraft. This form of combat can result from pre-warning of the enemy aircraft and/or armament which is essentially short range in nature; i.e. infrared seeking missiles or gun type weapons. Weapons of this type force the attacking aircraft to out-maneuver the opponent in order to attain a desirable tail position. In this area the target aircraft will attempt to shake off the attacking aircraft by diving to reach aerodynamic load factors in excess of those for the power limit. For a short period the need is for the highest g and therefor the smallest radius of turn possible at a given speed and altitude. Thus, a hard maneuvering relatively long duel will result. To engage in such combat the pilot must rapidly acquire the opposing machine visually during the fight - thus, a visual limit must be imposed. Choosing this limit as 6,000 ft turning radius; one nautical mile, we can compute and overplot this radius as line B on Fig. 2. In the region below this line, sustained turns equal to or less than 6,000 ft radius can be maintained. The aerodynamic limit of the aircraft would permit this region to be extended to line c if thrust was not a limit. The energy maneuverability technique can be used to examine the general effect of wing loading and thrust loading on the maneuverability of the aircraft. Fig. 3 compares the $P_s = 0$ contours for two fighter aircraft with significantly different wing loadings. This plot indicates a definite enlargement of the combat envelope bounded by the 6,000 ft constant state visibility limit for the light wing loading over the heavier aircraft. The thrust levels of these two machines has been adjusted to provide equal values of thrust loading for each.

A similar comparison for different thrust loading at constant wing loading is presented in Fig. 4. In this case a given aircraft was used and thrust rating changed by adjusting thrust levels. Again the larger or superior envelope coexists with the higher thrust loading. However, it is to be noted that to make significant gains in the steady state combat arena; which is subsonic, large thrust increases are required.

The conclusion that can be drawn from these two curves is that the superior fighter for maneuverability purposes will emphasize the lightest possible wing loading and the highest possible thrust loading commensurate with constraints of fuel and accuracy equipment such as avionics required to perform the total penetration combat mission.

Fig. 5 shows the C_{LMax}, versus Mach number capability of this aircraft with onset of buffet indicated. If the aircraft experiences buffeting, the induced vibration may degrade the crews ability to track the target. Fig. 6 depicts the aerodynamic load factor for onset of buffet. Comparing the 6,000 ft radius of turn steady state limit on this plot with that in Fig. 2 it will be noted that a requirement to avoid buffeting further degrades the altitude/velocity envelope for sustained combat. Some of the current experimental techniques used to predict buffet onset are shown on Fig. 7.

The problem of developing airfoils and devices to delay buffet onset is presently the subject of considerable study and research in NASA and the USAF. A review of available test data indicates that cambered airfoils usually show a somewhat higher C_L for buffet onset than non-cambered airfoils of the same thickness ratio. Fig. 8 shows the results of flight tests conducted in 1968 to investigate the existence of possible favorable effects of flaps on the buffet onset of a USAF fighter bomber the F-105D. That this improvement is basically due to control of flow separation is supported by the finding that a drag reduction; expressing itself as a range increase, could be achieved by optimizing LE and TE flap positions. Fig. 9 shows the results of using various leading edge flap positions to reduce the drag of the F-105 aircraft. Drag reductions of the order of 12% were obtained with 8 deg. leading edge flap.

The importance of maximizing thrust to weight and the ratio of useable lift coefficient to wing loading has been shown. Increasing these ratios, unfortunately, can rapidly lead to very large engines, large fuel requirements and heavy structural weight all of which rapidly degrade combat radius of action and result in a very large aircraft. Possibly, structural designers will provide some relief to the weight spiral through the use of high strength to weight materials, such as boron composites, in our future aircraft. Considerable research is underway in the US on these and similar materials and we can expect their increased use in the future. In order to obtain large improvements in maneuverability and enlarge the transonic combat zone we must look to improved aerodynamic configurations as well as higher thrust to weight ratios. What advancements can we expect from our current research programs?

Considerable progress has been made in the last five years in the aerodynamic design of supersonic aircraft. Some of the latest design techniques to come from this research were described in a 1968 American Institute of Aeronautics and Astronautics Paper (1) by D. D. Baals, A. W. Robins and R. V. Harris, Jr. of the NASA, Langley Research Center. These advances have come about largely through the rapid development of computerized numerical techniques which permit the solution of wing and body shapes with low supersonic wave drag and optimized lift distributions for minimum induced drag. The electronic digital computer has made it possible to implement these design techniques and has literally started a revolution in aerodynamic design. Work is now in progress to extend some of these methods into the subsonic area (2). With further improvement, these new techniques may be used to optimize the complete aircraft configuration, including such components as inlets, nacelles, external stores, and exhaust nozzles.

Significant advances in supersonic wave drag optimization procedures have been recently made by Harris of NASA Langley based on the theoretical approach published by G. N. Ward in the United Kingdom in 1955 (3). The numerical technique developed by Harris provides a direct solution of the fuselage required for minimum wave drag at supersonic speeds for a complete configuration. This technique is illustrated in Fig. 10. The left plot shows the average cross-sectional area distribution for a typical complete aircraft configuration with a cylindrical fuselage compared to a body with minimum wave drag. The plot on the right shows the cross-sectional area for a fuselage designed by this technique to provide the minimum drag. The method considers any number of restraint points on the fuselage. The computer locates the restraint points on the average equivalent body area distribution for the complete configuration and solves for the minimum wave drag shape through the restraint points. The shaded region shows the area which must be added or substracted from the original cylindrical fuselage to define the optimum body shape. Some typical calculated results obtained are shown in Fig. 11. The lower bound curve shows the minimum drag that would be predicted using the same three restraint points shown on Fig. 10 for this complete configuration. Also shown is the drag with the cylindrical fuselage and two other cases, one fuselage design for $M = 1.5$ and one for $M = 2.5$.

Considerable progress has also been made in the development of numerical techniques to define wing planforms and camber distributions which will provide minimum drag due to lift at a given design lift coefficient (4). Usually the wing is divided up into an array of small elements (500 to 1000) wherein the surface slopes are specified and resulting lifting pressures are calculated by following a precise routine starting with the front segments and working toward the rear. These new techniques offer the capability of calculating the chordwise and spanwise load distribution for arbitrary planforms, with variable camber and twist or can, conversely, calculate the camber and twist distribution needed to realize a given load distribution. Thus, optimized loadings can be developed for specific design conditions and for wing planforms which are optimized for mission requirements. One very real benefit which these new computer techniques bring to the designer is the ability to examine many different wing geometries and to develop configurations which have small aerodynamic center movement and low trim moments at supersonic speeds.

The capability of integrating the wing and body for optimum aerodynamic characteristics is now available using the wave drag and drag due to lift computer programs. Recently, considerable progress has been made in extending the programs to include the effects of canard and horizontal tail surfaces. Another recent addition to the lifting-surface programs provides for optimization of the wing camber surface in the presence of interference flow fields due to nacelles or external stores. Robert Mack of NASA Langley, has recently developed a numerical procedure for determining the wing slope changes required to remove various fractions of the interference lift created by these flow fields and has examined the resulting increments in drag due to lift and pitching moment at zero lift (5).

These techniques offer the designer new, more efficient tools to properly integrate the complete configuration into an optimum aerodynamic shape. The capability for computer generated drawings provides additional flexibility to check for input errors to the numerical programs and to refine the configuration in local areas. Once the numerical model has been specified in sufficient detail

it becomes a simple step for the computer to print out component surface wetted areas, cross-sectional area cuts and reference lengths.

These refined design methods are very useful additional tools which permit rapid evaluation of the effects of many configuration variables on mission and operational capabilities. They serve to narrow the broad spectrum of configuration geometry to the favorable candidates. We can expect that increased computer capacity will soon make it practical to combine many separate programs into a computer complex which will analyze such factors as propulsion performance, structural design and weight, aeroelastic properties, and complete load distributions on specific aerodynamic configurations. Truly a revolution in aerodynamic design is occurring and the old component build-up methods are rapidly vanishing from the scene.

Another area in which we can expect improved aerodynamic design is in the development of optimized airfoil sections to increase drag divergence and buffet limits at high subsonic speeds. Much of the current research on improved airfoils is based on the early work of H. H. Pearcy in England (6). His basic idea was to contour the leading edge to rapidly expand the flow from the stagnation point and generate supersonic speeds over the nose region. The expansion waves created are reflected as a series of compression waves from the sonic line. The compression waves gradually reduce the local flow Mach number and decrease the shock and separation losses on the upper surface. The resulting pressure distribution has a marked peak near the leading edge which gave the name "Peaky Airfoil" to this design.

Another approach has been under development by Dr. Whitcomb at NASA. He has developed a camber distribution which reduces the adverse pressure gradients over the upper surface which are due to angle of attack. His airfoil is contoured to carry more of the lift over the rear portion and less over the forward section. Wind tunnel tests have been quite favorable and flight test of a US Navy F-8 fighter with a new wing designed by Dr. Whitcomb are planned in the next 2 or 3 years.

Although these new airfoil sections have been designed primarily for wings of moderate thickness ratio, such as used on subsonic transport aircraft, there is a good possibility that the same principals will apply to thinner supersonic wings. In this case the desired chordwise lift distribution will be obtained by using leading and trailing edge flaps to provide the right amount of camber at each flight condition. The use of boundary layer control or a jet flap also appears quite promising to reduce the shock losses at high subsonic and transonic speeds.

The digital computer has become an essential tool to develop optimum aerodynamic configurations, but the wind tunnel will remain the real workhorse for the designer within the foreseeable future. In fact, we have seen greatly increased use of high subsonic, transonic and supersonic tunnels on our new designs. The tailoring of any new configuration requires extensive testing since the successful integration of the complete configuration, airframe, inlet and exhaust nozzles is beyond the capabilities of our current generation of computers. In addition, analytical methods are not now available to handle the complex flow interactions in the vicinity of the inlet and exhaust nozzles. Airframe-propulsion compatibility and integration remains as one of our most complex problems but one which must be completely solved to achieve an optimum fighter design.

3. STABILITY AND CONTROL

The preceding sections of this paper have been devoted to those design characteristics which are necessary in a fighter design to provide superior performance, range, speed, climb and maneuvering ability. It is now necessary to discuss the equally important requirements from the standpoint of stability and control. A discussion of stability and control must include a discussion of flying qualities. Stability and control is such a major contributor to flying qualities that both must be included in a discussion of either one. Other disciplines which also affect flying qualities such as flight control, instruments, display, and structures will not be included here. Some discussion of the use and impact of stability augmentation is presented but only with the view of its relationship to flying qualities; not as to design techniques or design philosophy.

When flying qualities problems occur that are basically due to stability and control deficiencies they are frequently treated or overpowered by working on one of the other technical disciplines since it may be more expedient to do that than to correct the basic deficiency. As a result, there has not been an appreciable advance in the pure stability and control world of derivatives, areas, and moments which configuration variations such as shape, size and placement of aircraft components will affect. If anything the solution of flying qualities problems by stability and control means has been avoided or sublimated to satisfy the demands of more performance and less weight. It must be left to the reader's experience to judge what the failure to fully identify and solve the real problems has done to system complexity, reliability, and maintainability in addition to the situations encountered in flight.

The air superiority fighter must be efficient over a wide range of speeds and altitudes. Having the capability of being highly maneuverable, or the ability to reach and maintain high load factors and high angles of attack, is not sufficient in itself. The aircraft must be able to respond to the pilot's needs. It must be a steady, or stable, aiming platform, it must readily change attitude to respond to the meneuvers of the target or to avoid becoming an easy target. In other words, it is not enough for the airplane just to get to the fighting area and situation. It also must do something and do it well once it is there to be an effective weapon. Stability and control is the means by which these demands are met and the means by which weapon effectiveness often depends.

As obvious as these things are they are frequently disregarded early in the design due to the emphasis on performance. Good flying qualities are assumed during the early stages at just the time when the aerodynamic designs, which could assure them, are being prepared under the low drag-high performance syndrome. In addition, during the design phase, the concentration has been on level flight capabilities in calm air, i.e., the ideal world, with the tests and analysis concentrated on this type of flight. Due to this concentration it is not until operational and service experience are obtained that it is found that stability and control problems exist which are serious enough to affect the safety as well as the usability of an airplane. When such situations occur the emphasis is no longer on performance and the user gladly gives up a few knots of speed or miles of range in exchange for a safe and usable airplane. A balanced design which considers all the flight conditions in which operation is intended and has the full confidence of the pilot will be more useful and successful than many past aircraft.

Since the primary method of changing the stability characteristics of today's aircraft is by the use of stability augmentation systems it is proper to mention this area first. By stability augmentation systems we mean the equipment which changes the "bare airframe" characteristics by means of automatic feedback systems. In some cases this trend has gone too far and the pilot, the airplane, and the mission, would all be better served if more bare airframe stability were incorporated into the design. We are not, however, advocating the elimination of stability augmentation systems. Augmentation can often do a necessary job efficiently and with less penalty than can some other means. The whole point is to arrive at a better balance between airframe and augmentation. The remainder of this section of the paper will discuss some of the stability and control aspects that should be considered when arriving at the balanced design.

The augmentation systems in use today are more extensive in operation, complexity, and number than their original counterparts of 20 years ago. Their function, however, is primarily the same, i.e., improve stability and control characteristics. Also unchanged is the flight region in which this improvement is to occur and where design emphasis is placed. Stability augmentation systems are primarily designed to improve the airplane's characteristics in level flight or in the landing approach. The analyses are conducted for steady equilibrium conditions of flight. The most frequent conditions which determine augmentation system design are straight and level flight, constant altitude turns, and rolls at constant rate. There is some sense in this for even fighter aircraft spend a considerable portion of their time in or near level flight and then go through a landing approach at least once per flight. They also turn and roll though not often at constant altitude or constant rate. But fighter missions are not intended to be conducted primarily in level flight or constant anything and in the course of training and combat there are a great many unusual attitudes or maneuvers encountered which are not fully considered during augmentation design. The result of not considering extreme, but often necessary, maneuvers is either an unsatisfactory airplane or a restricted airplane; neither situation is acceptable. Often, during such maneuvers the augmentation system will oppose the pilot's commands or provide a control motion that will cause the airplane to get into trouble.

The stall and spin situation is one area where the augmentation system can provide problems. There are current fighter aircraft where the pilot is instructed to turn off the roll damper at high angles of attack to aid in preventing spin entry and to insure spin recovery. This occurs because the roll damper attempts to hold the wings level during the stall which results in frequent aileron motions. In addition to a disconcerting wing rock, the continued, frequent aileron deflections on a swept wing aircraft may induce adverse yaw and pro-spin yawing moments. To avoid this the pilot must remember to turn off the roll damper at high angle of attack. In training or in combat this situation is clearly unsatisfactory since he will be very busy doing other things relating to the mission and should not be expected to perform this switching at critical moments. Of course he must also remember to turn the damper on once he recovers from the maneuver or returns to level flight. During spins augmentation systems have resisted pilot recovery efforts by opposing the pilot control inputs and reducing the control available for recovery. In some aircraft this can prevent the recovery. To avoid this, pilots are instructed to turn off all augmentation before attempting recovery.

Other non-level flight maneuvers have been hampered by stability augmentation. Some of the tactics taught to fighter pilots require steep climbing turns and such maneuvers are known to require the pilot to use a significant amount of aileron to counteract the roll damper input which is opposing the steady state rolling motion of the turn. As a result of this and other problems involving the airplane-pilot-augmentation system, it has been determined through interviews that veteran fighter pilots frequently turn off all of their augmentation devices when in combat or combat training. This is an ironic situation since the augmentation is provided to improve the airplane and give the pilot a better combat vehicle whereas the pilots actually feel that it detracts from the airplane's capabilities during combat. This is due, again, to the fact that augmentation is not designed with consideration of the maneuvers and attitudes of combat situations. True, these attitudes are encountered a small percentage of the flight time, but these are the times the airplane is doing its intended mission so the importance of this percentage is quite high.

The obvious conclusion here is that augmentation systems are not doing their intended job and often detract from or worsen the unaugmented or bare airframe characteristics. Since the bare airframe characteristics themselves have been judged inadequate so that augmentation is necessary, the pilot is not being provided with the airplane that he should have. A great deal of work is needed here if augmentation is to do what it is supposed to do and do it at all flight conditions. In any event we should always examine these extreme flight conditions to determine how to provide the needed flying qualities which may not always be most efficiently obtained via augmentation.

It has probably been noted that all the problems with augmentation systems discussed above

occurred at low speed and it may be said that a fighter, with propulsion, configuration, structure, and systems all emphasizing transonic and supersonic speeds should not be compromised by low speed problems. In actual fact, low speed characteristics and capabilities are usually the key to a successful fighter. Regardless of the speed at which combat starts continued maneuvering and frequent high load factor exposure quickly result in the aircraft arriving at low speed flight. The same is true of air to ground weapon delivery where repeated delivery passes are made and the aircraft must maneuver in the target zone to avoid losing the target. Low speed maneuvering is emphasized in training so low speeds are encountered frequently in the modern fighter. Stall warning speeds and unfortunately stalls are not an uncommon occurrence on our fighters today so it is extremely pertinent to consider the low speed regime and to rank it at least equal in importance to the high speed areas which already receive so much emphasis.

Even though low speeds and extreme maneuvers and attitudes are emphasized in training and in operation they are not emphasized in design. Wind tunnel testing is seldom done past the stall angle of attack and even less often are combined high angle of attack-high angle of sideslip data obtained. It should be a standard part of any new fighter plane design and development program to obtain wind tunnel stability and control data at angles of attack up to 90^o and at sideslip angles up to 45^o . These data should then be used to determine stall, post stall, spin, and spin recovery characteristics. These large amplitude, non-linear situations can now be attacked for analysis purposes by the computer so the data can be used. The need for such analysis is established due to the frequent occurrence of low speed maneuvers near or at the stall, the more than occasional encounter with the spin and the difficulties encountered with spin recovery.

When such characteristics are determined perhaps it will prevent the acceptance of such high angle of attack characteristics as those shown in Fig. 12. Note that the directional stability parameter, C_{N_β}, becomes unstable and the lateral stability parameter, C_{L_β}, experiences a noticeable reduction at close to the stall angle of attack. These characteristics were not discovered until after several years of flight experience and many reported stall problems required an extensive examination of the airplane's aerodynamics.

In addition to providing the data necessary to analyze the low speed characteristics where so many problems have occurred and where fighter aircraft are repeatedly operated, this information will also make it possible to make decisions on augmentation systems. As was pointed out, present augmentation systems are not designed to cope with low speed, large attitude flight. Perhaps it would not be efficient to have augmentation do this job if the control authorities required plus the added weight and complexity were added to those already used by augmentation system. It may turn out, if sufficient emphasis is placed on the problem, that aerodynamic or basic airframe changes are the best way of obtaining good stability and control characteristics. There have been some initial indications that this might be true but these are so preliminary and so limited that this should not be taken as a conclusion. However, it is worthy of detailed examination to determine the design possibilities.

The jet engine has provided the fighter aircraft with a load carrying capacity much greater than the airplane can efficiently absorb internally. The result can be seen on fighter aircraft throughout the world by noting the vast number of external stores that are carried. But stability and control characteristics during the design process are still being determined primarily by considering a clean airplane, i.e., one without external stores. Many of these stores, either due to their shape or their location on the wing, result in severe stability and control problems (Fig. 13). An earlier consideration of stores and their effects is needed - needed during design. Certainly, it is neither feasible or possible to consider all stores and their combinations for many stores are designed after an airplane is in service. But some consideration of representative stores and loadings is desirable and possible. This may affect the balance of the aircraft and the design of the flight control system even if no basic configuration and stability changes are needed. Certainly, it is not sensible to ignore the military payloads during the design.

Another external store design consideration is that of store asymmetry. It is not uncommon to drop stores in such a way as to result in temporary asymmetries until another pass can be made. There have been situations where ejector rack failures have resulted in unintentional asymmetries. Frequently, these situations result in severe control problems because they were not given proper attention during the design phase.

Early design consideration must be given to the interaction between stability and control and structures. There are too many instances of severe structural problems developing as the result of stability and control problems or deficiencies. These stability deficiencies have allowed aircraft to attain flight attitudes that were outside the structural design envelope. Structural failures have occurred due to pitch-up, inertia coupling, and directional instability at high Mach numbers. When such problems occur they either result in added structural weight to withstand the loads or in restrictions in speed and maneuverability, or both, which is undesirable.

One fighter aircraft with a T-Tail is stable with increasing angle of attack until the horizontal tail encounters wing downwash. The airplane then becomes unstable as shown in Fig. 14. This problem was encountered on an instrumented flight test during an abrupt pull-up to 80% of the design limit load factor; however, the airplane pitched up and experienced loads in excess of the 80% limit. Later analyses showed that the airplane could go unstable at high angles of attack and that the resulting structural loads would destroy it. Even though some control fixes were added strengthening of the forward fuselage was required to absorb the added loads.

What to do about turbulence is still not clear. A fighter plane with its high load factor capa-

bilities should not suffer from simple aeroeleatic deflections from turbulence so the stability derivatives should not be degraded due to aeroelasticity. But there are other effects and these should be investigated. Exactly how to make the investigations and the specific levels and details of the turbulence that should be used are not known at this time. It is clear that turbulence can change the rigid body characteristics and can result in augmentation saturation. Therefore, the designer cannot fully complete his job without some thought and analysis of what turbulence will do to his airplane. Similar analyses are already required for the structural design and the need is equally applicable to the stability and control designs. It is the same turbulence and even though it is looked at from a different point of view the goals are the same.

The problems and some short comings of stability augmentation systems have been discussed previously and whether or not augmentation is the best way to solve these problems was mentioned. It was just low speed problems that were being considered when it was noted that perhaps airframe stability should be given more consideration. This is not to say that we do not believe in closed loop type augmentation devices. We feel that augmentation performs an essential and valuable function for the supersonic, wide altitude range fighter. But again, the design must be one of balance between airframe stability and control and augmantation stability and control and in obtaining the balance the designs must fully consider the effects of failures.

The problem of failure has been with us for a little more than 20 years, or since the beginning of the use of augmentation. Hard over failures have, after some agony, been pretty well eliminated by redundancy but passive failures, partial failures, and complete failures are still with us and probably always will be; hopefully, to a lesser degree with time. Decisions are extremely difficult in this area for trade-offs in drag and weight effects on performance must be made against complexity, maintenance problems, and failure situations. Augmentation systems which do provide against disastrous failures and continue to be effective after some failures are necessarily sophisticated and complex. Complexity leads to maintenance problems and problems of getting the equipment to work properly and keeping it working in the operational environment. Historically, these devices never seem to fulfill the reliability predictions made for them; however, this is not the place to become involved in reliability prediction arguments. The designer must be aware that if a failure can occur it will occur and some design attention must be given to the analysis of all possible failures. This has been done quite successfully in several instances in the U.S. and we are striving to see that it continues. The emphasis to be placed on all various failures is the designer's dilemma. Some will be so remote as to receive almost no consideration. Others will be more probable and must be capable of being overcome by one means or another if the mission is to be continued. The great mass of failures is usually between these two extremes and each must be thoroughly examined not only to determine its effect on the stability and control of the airplane but to also ascertain its effect in conjunction with other failures. Again, as has been stated many times before, the extreme flight conditions should be thoroughly examined along with the level flight when assessing failure effects.

The recent total revision to the U.S. military flying qualities specification MIL F-8785B (Ref.7) treats some of these problem areas. The total mission is separated into flight phases where a particular general task or objective will take place such as weapons delivery or cruise or power approach. This enables the designer to look at individual portions of the mission and emphasize those that are important to his airplane. For the strike fighter, the air-to-air combat and ground attack phases would be the most important, for that is the job the aircraft is designed to do. Also, to be included when tabulating the phases are take-off, climb, cruise, approach, and perhaps in-flight refueling in addition to any others that might be required for a given airplane.

The flight phases are grouped into three categories in order to combine them into families of like difficulty and precision. One category lists those phases in which the most precise and difficult flying will occur and are usually the phases which include the basic function of the airplane. These are all non-terminal tasks and include such phases as combat, weapon delivery and anti-submarine search. The next category is also non-terminal and groups those tasks that are required to go to and return from the basic mission location such as climb, cruise and descent. Pilot tasks in this category are usually gradual and do not require high precision. The third category of phases are those accomplished in the vicinity of the airport and include takeoff, landing, and go-around.

For each of the flight phases which combine to make up the whole mission of the aircraft it is required that an envelope be prepared which defines the boundaries of speed, altitude and load factor that will be encountered in that flight phase. This, then defines the environment within which the plane will be flown. This individual flight phase envelope idea also places emphasis on the more important areas of use because the usually mundane phases of climb, cruise and let down are separated from combat or prime use phases. Each flight phase can then be examined individually in order to fully consider all tasks and maneuvers that must be accomplished within the phase.

Additional emphasis is placed on the primary flight areas by having three different envelopes; operational, service, and permissible. The operational envelope bounds the speeds, altitudes, and load factors in which the airplane must be capable of operating to accomplish its design missions. This is where the airplane is meant to perform. This is where it has to be at its best and for some flight phases the envelope should be as large as possible. Remember that each flight phase of the overall mission must have these three envelopes to permit a detailed examination of all parts of the mission and to allow the requirements to apply where they are most needed and to be applicable to the tasks that must be performed in each phase.

It is realized that the operational envelope will probably not include the full scope of the airplane's capabilities. The service envelope, which is to be coincident with or outside the

operational envelope, is to define airplane limits rather than mission requirements. The airplane can and will fly here but it is not intended to fly in these regions while accomplishing its mission, therefore, the flying qualities do not have to be as good as those for the operational envelope. The service envelope will include the extreme regions of altitude, speed, and load factor; regions that are encountered less often than those of the operational envelope. By reducing the requirements in these regions the designer does not have to add equipment to provide a high level of flying qualities for an infrequent situation which would penalize the aircraft when performing its principal function.

There may be still more area outside at least certain portions of the service envelope where flight is both possible and permissible. We are referring to flights where stalls, spins, zoom climbs, etc., are encountered and a third envelope called the permissible envelope has been added to include such flight. The likelihood of occurrence is even lower than that of the service envelope and flight in this envelope is considered to be transient in nature. No quantitative requirements are imposed in the permissible envelope since there are areas-stall and spin for example, where it is not possible to achieve a stabilized flight condition. Our new requirements specify that such flight shall be safe and it shall be possible for the pilot to easily return to the service envelope.

This use of the multiple flight phases and the defined envelopes focuses attention to the intended areas of use of an airplane. This theme is further carried out by levels of flying qualities which specify a minimum value of a particular handling quality parameter which are related to the ability to complete the mission for which the airplane is designed. There are three levels which are; 1. Clearly adequate; 2. Adequate but with some increase in pilot workload or decrease in mission effectiveness; 3. Safe but with excessive workload or inadequate mission effectiveness. The actual requirements specified by the levels may differ between flight phase categories so that values for a given level that apply to an air-to-air combat situation are not the same as the values for the same level for cruise or for the approach. This again allows each situation to be examined based on its own requirements to tailor the requirement to the need rather than specifying a single value for all flight conditions which may be applicable at only one flight condition.

From the stability and control engineer's viewpoint it would be nice to have the best level of flying qualities at all times but this is clearly impractical because of the penalties that would result. But we do believe it is essential to have the best level of flying qualities, i.e., clearly adequate, a high percentage of the time in the operational flight envelopes since that is where the missions are intended to be flown. Since this area is so important the first level, or best level, of flying qualities is required. In the service envelope it is permissible to allow the pilot to work harder and have a less effective airplane because such flight is not directly related to the mission success. The second level of flying qualities is required in the service envelope. The best level may not use the same quantitative values for each of the categories since clearly adequate for cruise does not have to be the same as does clearly adequate for air-to-air combat.

What happens or what is required in the various phases following a failure? For that matter what failures must be considered; only single failures or should double failures also be included? When looking at this problem it was found that we could not answer this question of single vs double failures. Some multiple failures, double or triple in nature, occur more frequently than some single failures so it did not seem reasonable to specify all single failures and exclude all double failures. How far to go into multiple failures was not directly answerable either. It was possible to answer what failures should be considered by saying all failures that affect flying qualities. This goes beyond the realm of stability augmentation and includes instruments, sensors, control system components propulsion system and perhaps the fuel tank system.

Since the number of failures was not a reasonable determination and since all pertinent failures had to be considered it was decided to relate the requirements to the probability of encountering a failure condition. It is now required that the probability of occurrence of each possible failure be determined and also the effect of this failure on flying qualities. By combining this information the overall probability that one or more of the specified flying qualities are degraded to the second or third level can be determined.

The requirements are that the probability of encountering the second level of flying qualities within the operational envelope should be no greater than once per 100 flights and the probability of encountering the third, or just safe, level of flying qualities should be not greater than once per 10,000 flights. For flight in the service envelope the required second level capability should not degrade to the third level more often than once per 100 flights as a result of failure.

No failure or combination of failures is permitted to degrade any of the flying qualities below the third level. For the airplane which relies heavily on stability augmentation to achieve the best level of flying qualities the third level is sometimes thought of as bare airframe stability requirements. The actual requirements do not so state this and there may be, and often are, good reasons for having the bare airframe meet the second or first level in at least some areas. So the lowest level should not be considered to be specifically intended to apply to the unaugmented airplane for, as discussed previously, the balanced design will look for the best way to obtain good flying qualities (the first level) and this does not necessarily mean by augmentation. By having this requirement, however, which does not permit flying qualities to degrade below the third level, a limit is established for complete augmentation failure. It was not intended for the designer to just obtain the third level from the bare airframe.

There is nothing in the specification to assure that all the maneuvers and unusual attitudes are

examined. This area still must be carried out by discussion and emphasis on the use of the airplane. It should be readily apparent that any fighter will be highly maneuverable and that the extremes encountered in a maneuver should be a design consideration and not an afterthought that the pilot reports on when it is usually too late to do something about the real problem.

When external stores must be carried they are considered to be a normal part of the configuration and the regular requirements apply. There is no reason why the requirements should be relaxed if stores are a necessary part of allowing the airplane to complete its mission. If a given flying quality is necessary in the flight phase in order to get the job done it is needed with the stores as well as without so no relaxation is justified.

The problem of how to account for atmospheric turbulence has been approached but not solved. A continuous random turbulence model and a discrete turbulence model are specified to be used in the analysis to determine the effect of turbulence on the flying qualities and ability of a pilot to recover from the effects of discrete gusts. This is just an approach to the problem because we know that any turbulence model can be shown to be wrong under some circumstances. We do not know enough about what the flying qualities should be in turbulence to prepare a thorough set of requirements and this must wait for additional research to provide these answers. What has been done is to make a start and require the designer to analyze the stability and control characteristics in turbulence to determine if the control system saturates and to determine if structural mode problems are present.

4. CONCLUSION

The air superiority fighter must have exceptional maneuverability in the subsonic and transonic regions to achieve maximum effectiveness. Aerodynamic design improvements which reduce the total drag and buffeting at high lift coefficients are likely to come from current programs and should provide significant improvements in maneuverability and performance. The rapid development of numerical solution techniques using the digital computer gives the designer improved analysis methods with which to optimize and select optimum aerodynamic configurations.

Insufficient attention to the high angle of attack flight regime during design has resulted in stability and control and flying qualities deficiencies. Stability augmentation system design has not overcome these problems due to a similar lack of design attention in this area. In some cases, the autmentation system has introduced undesirable control surface motions which have made the aircraft more difficult to control. The new and completely revised US military flying qualities specification, MIL-F-8785B, lists flight phases descriptive of most military missions and provides the necessary flying qualities based on the flight phases and mission requirements. All subsystem failures which affect flying qualities must be determined and the probability of occurrence of each failure and its impact of flying qualities presented. The requirements place increased emphasis on those regions that are most important to mission success.

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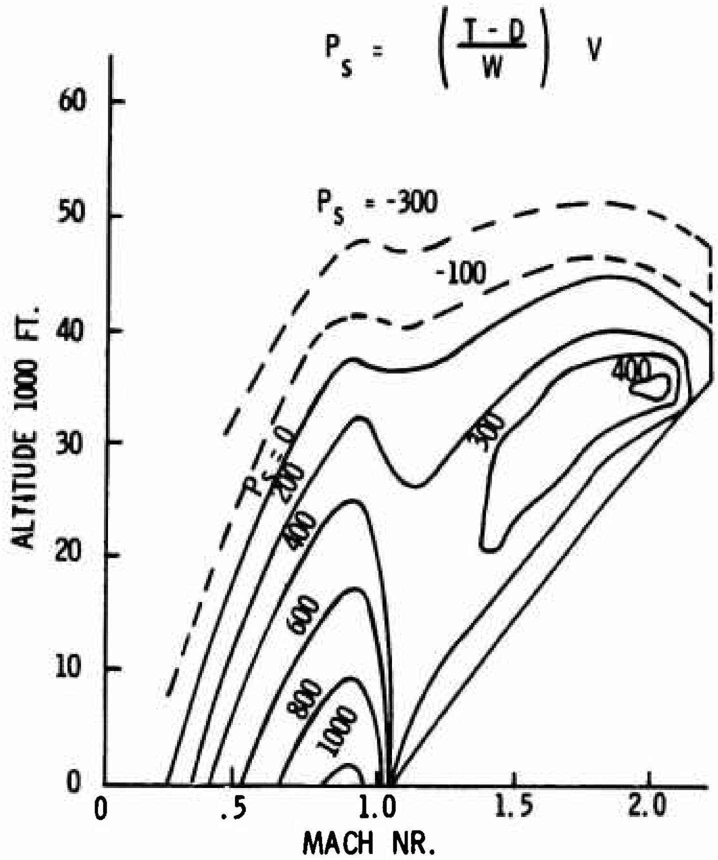

Fig. 1 3g Energy Maneuverability Limits

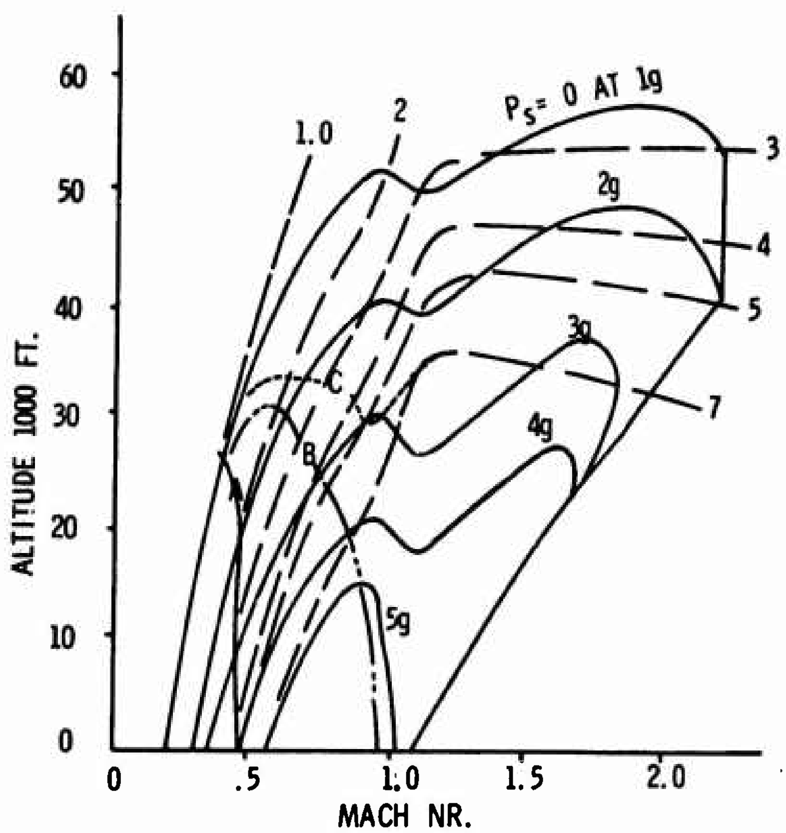

Fig. 2 Limiting Parameters

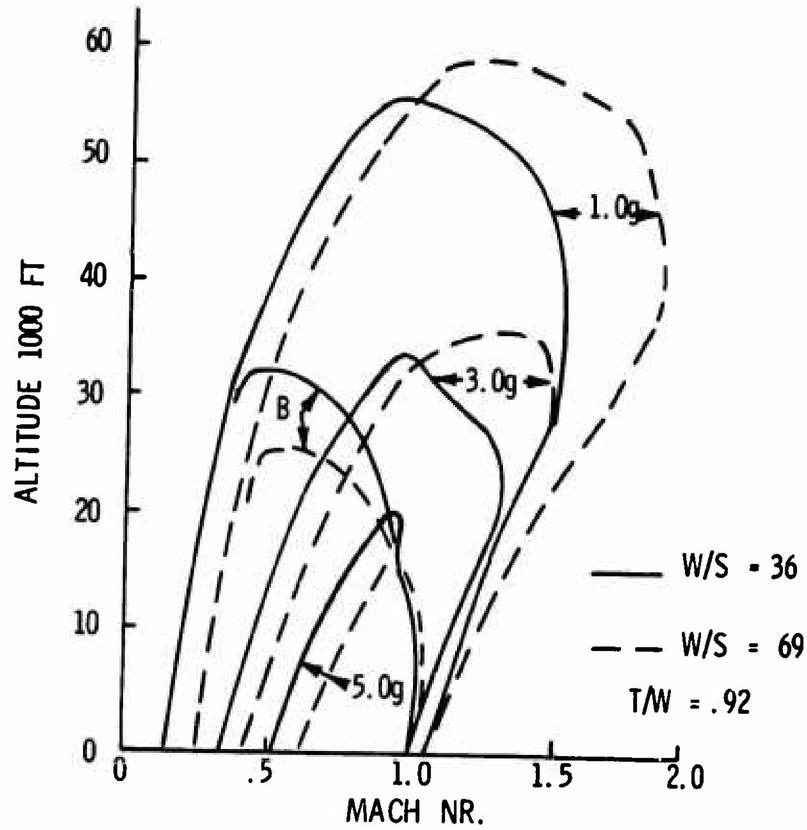

Fig. 3 Wing Loading Effect

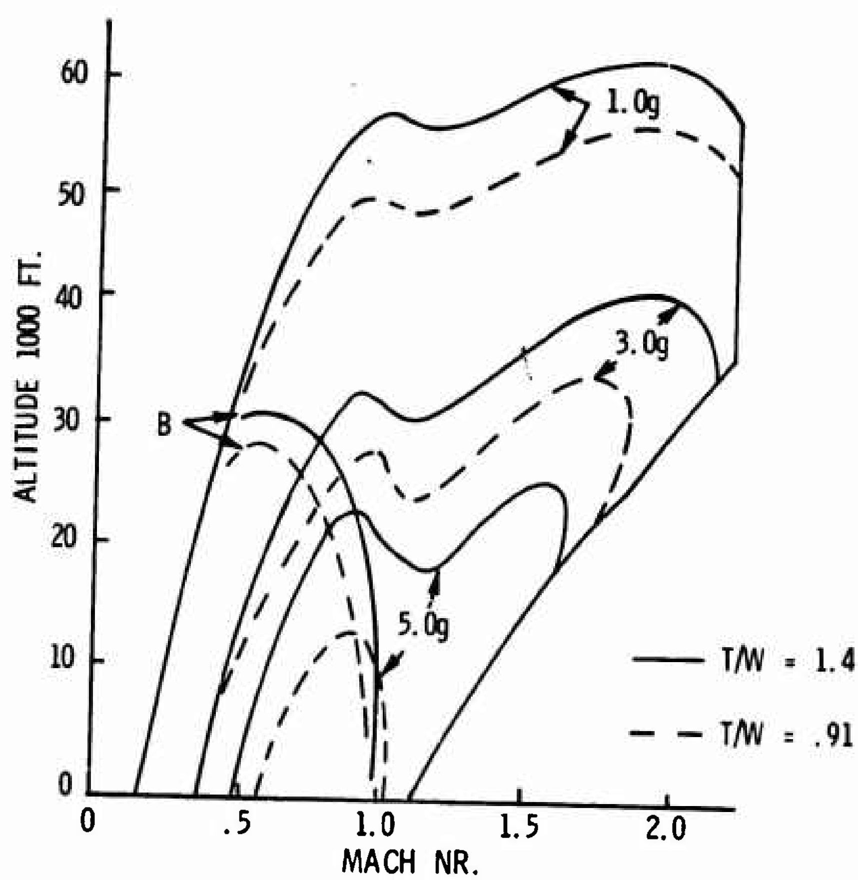

Fig. 4 Thrust Loading Effect

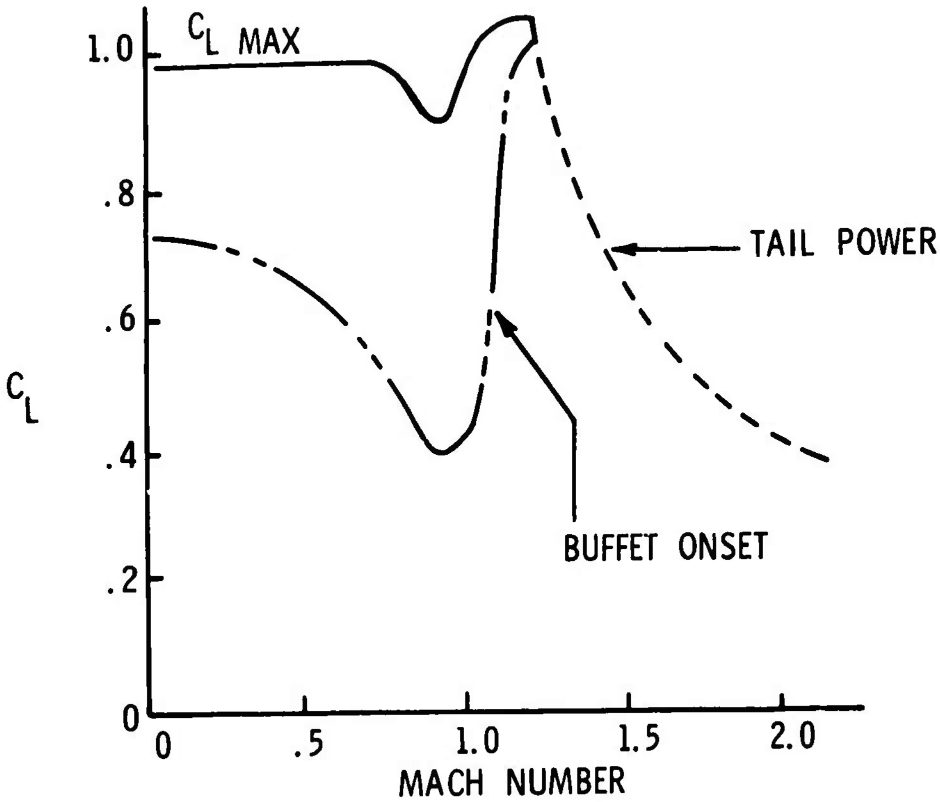

Fig. 5 Maneuver Limits

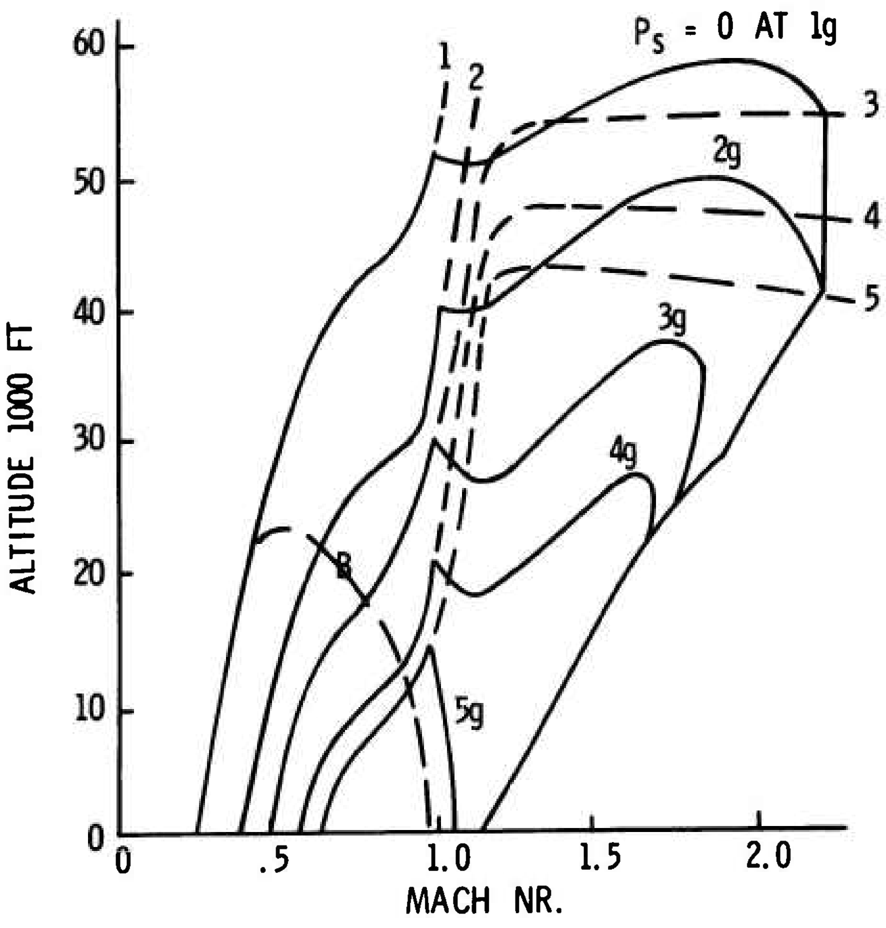

Fig. 6 Buffet Onset

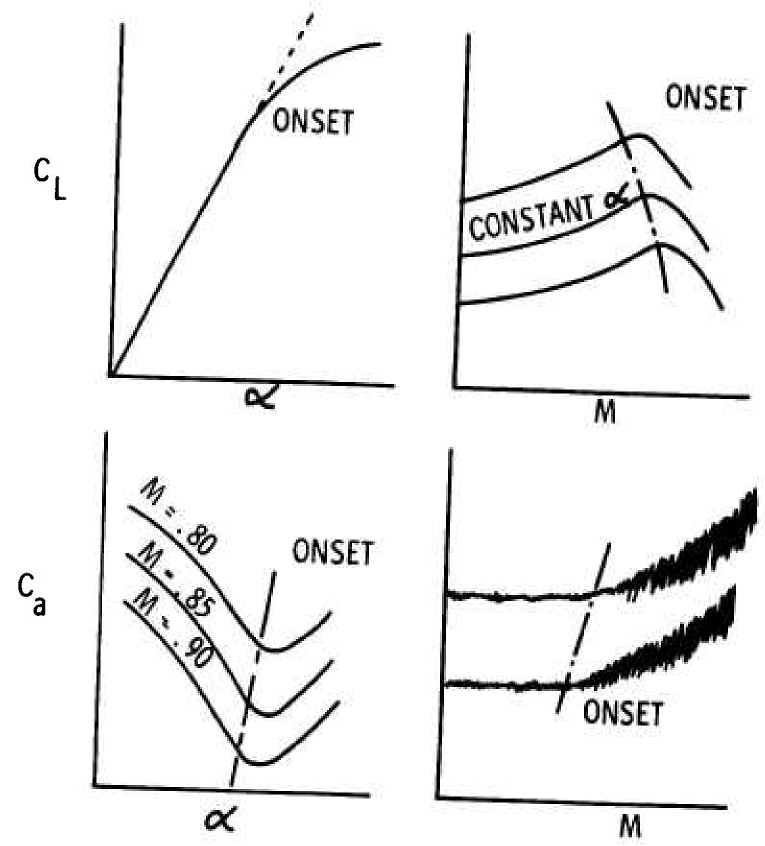

Fig. 7 Buffet Onset Prediction

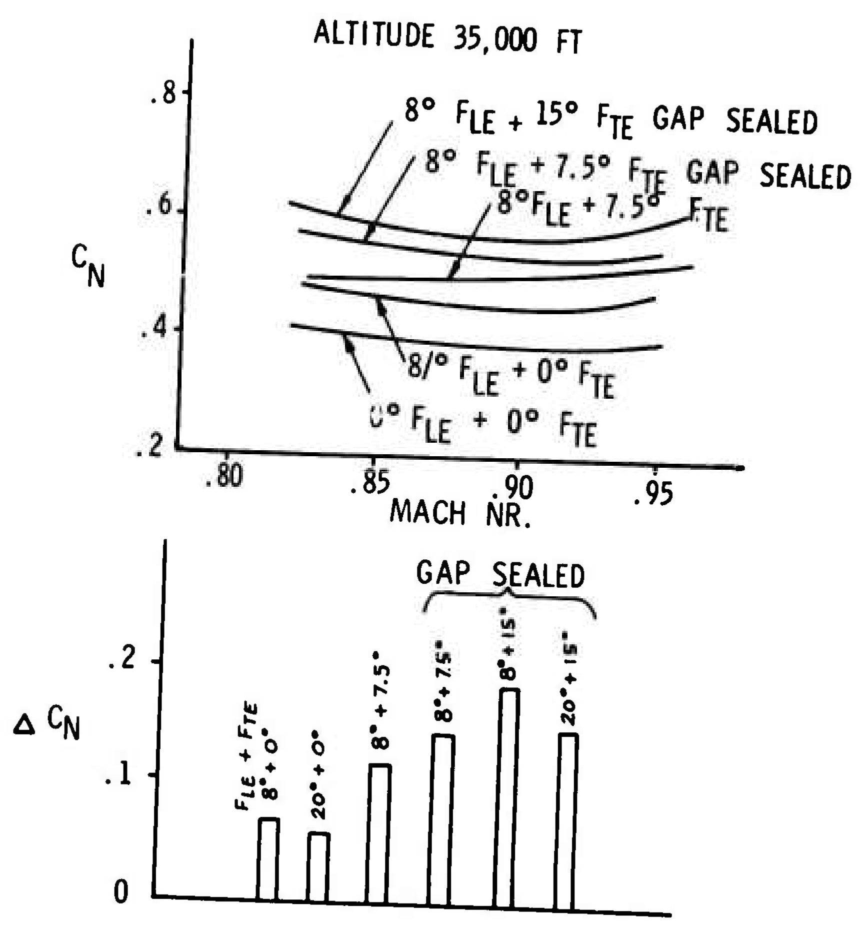

Fig. 8 F105D Buffet Onset

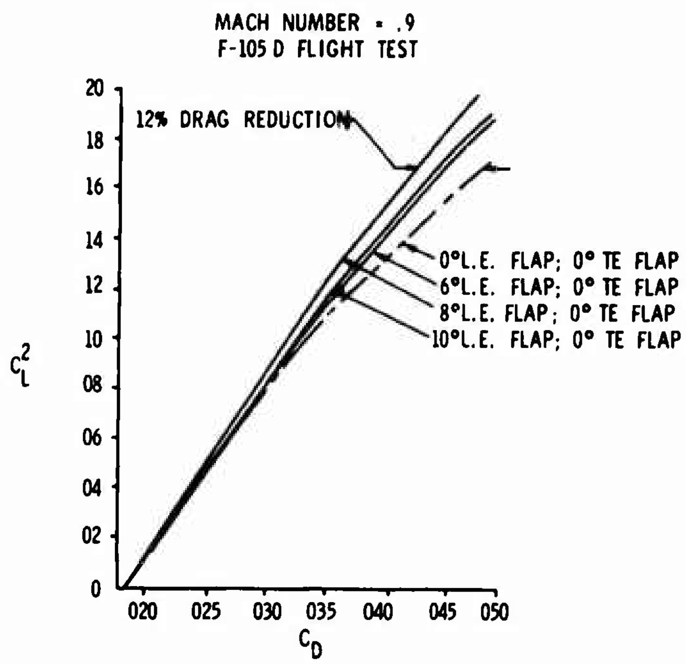

Fig. 9 Effect of Leading Edge Flap

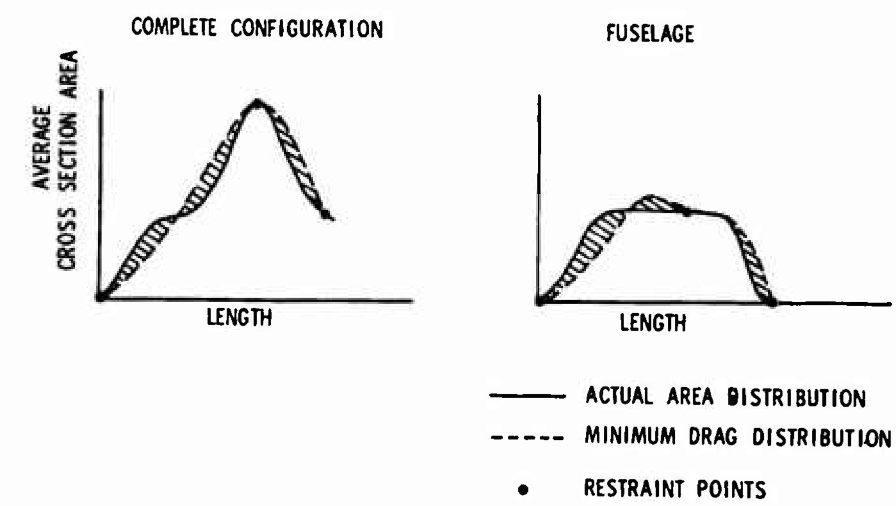

Fig. 10 Fuselage for Minimum Wave Drag

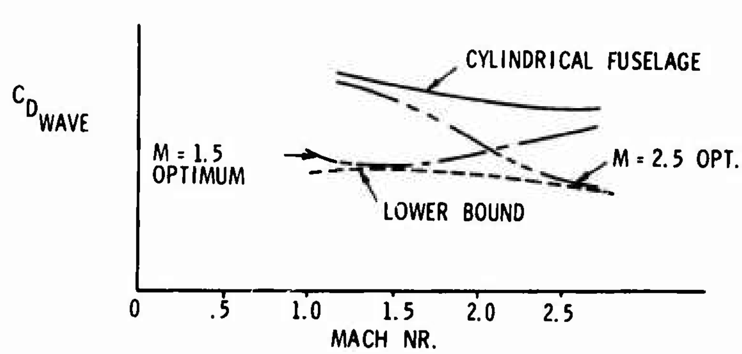

Fig. 11 Effects of Fuselage Optimization

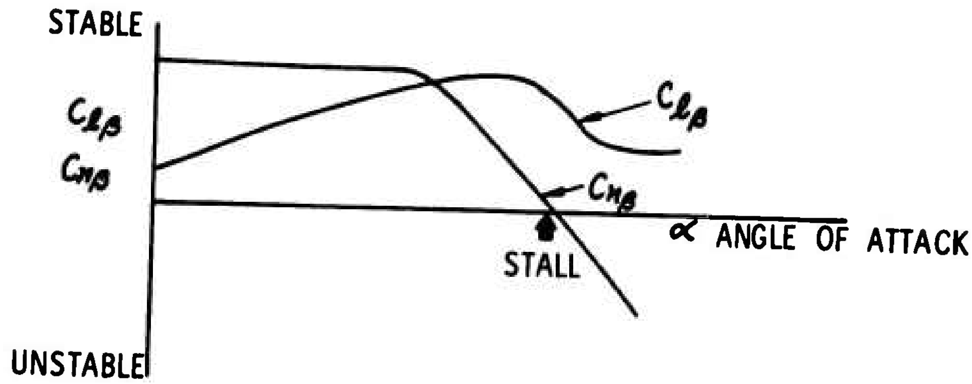

Fig. 12 Lateral & Directional Stability

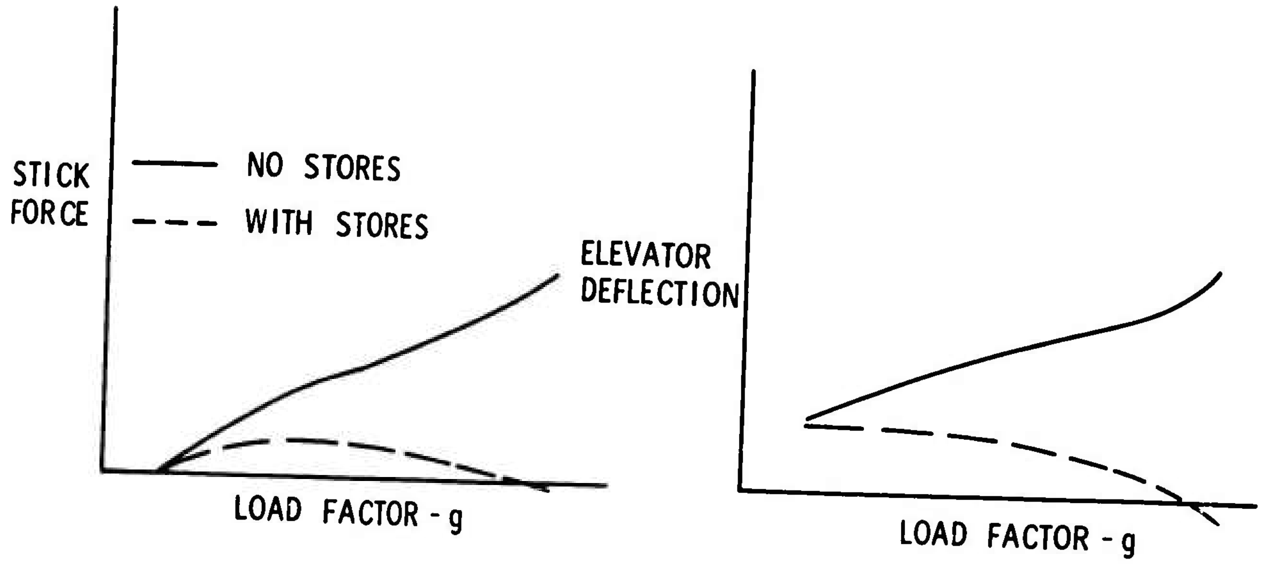

Fig. 13 Effect of External Stores on Longitudinal Maneuvering

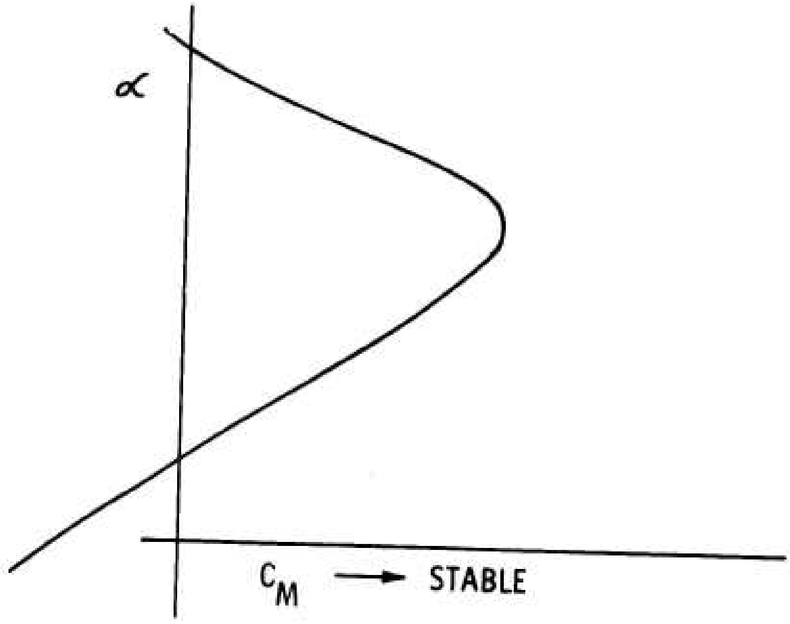

Fig. 14 Longitudinal Instability of Pre-Stall Angle of Attack

DESIGNER'S VIEW OF POWERPLANT PROBLEMS

by

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PRELIMINARY DESIGN ASPECTS OF MILITARY AIRCRAFT

DESIGNER'S VIEW OF POWERPLANT PROBLEMS

1.0 SCOPE

I shall interpret this title as applying to the problems of installing the powerplant in the airframe; this is taken to comprise firstly the overall problem of layout and secondly the detailed problems arising at the engine/airframe interface. Thus, I shall touch only briefly on the internal machinery of the powerplant, about which we may hear from later speakers, and I will leave the field of supersonic intake design to the last of this morning's lectures. Furthermore, in order to restrict this presentation to 35 minutes, I shall limit my discussion to the jet engine as installed in fighter type aircraft.

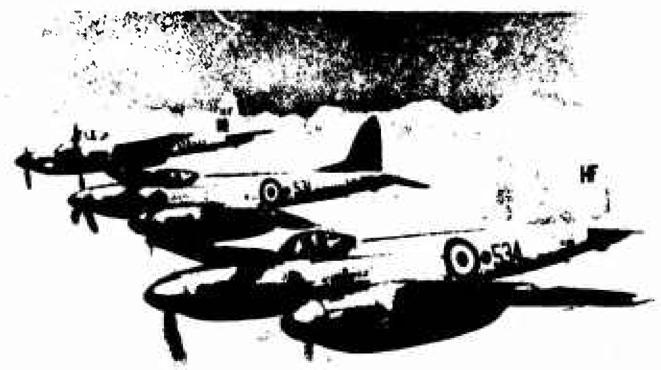
2.0 PRELIMINARY DESIGN

'Preliminary design' is taken to be the period of iterative testing of major design compromises which is required in order to limit the number of solutions which go forward for deeper study. Several returns may be made to this early stage before main-line design commences.

Preliminary design is a forward-looking process but we may examine its outcome in the successful aircraft types of the past. I will, therefore, trace briefly the development of the jet fighter back to the end of the piston engined era; using as illustration some of the fighter aircraft produced by Hawker Siddeley Aviation.

3.0 DERIVATION OF THE JET FIGHTER

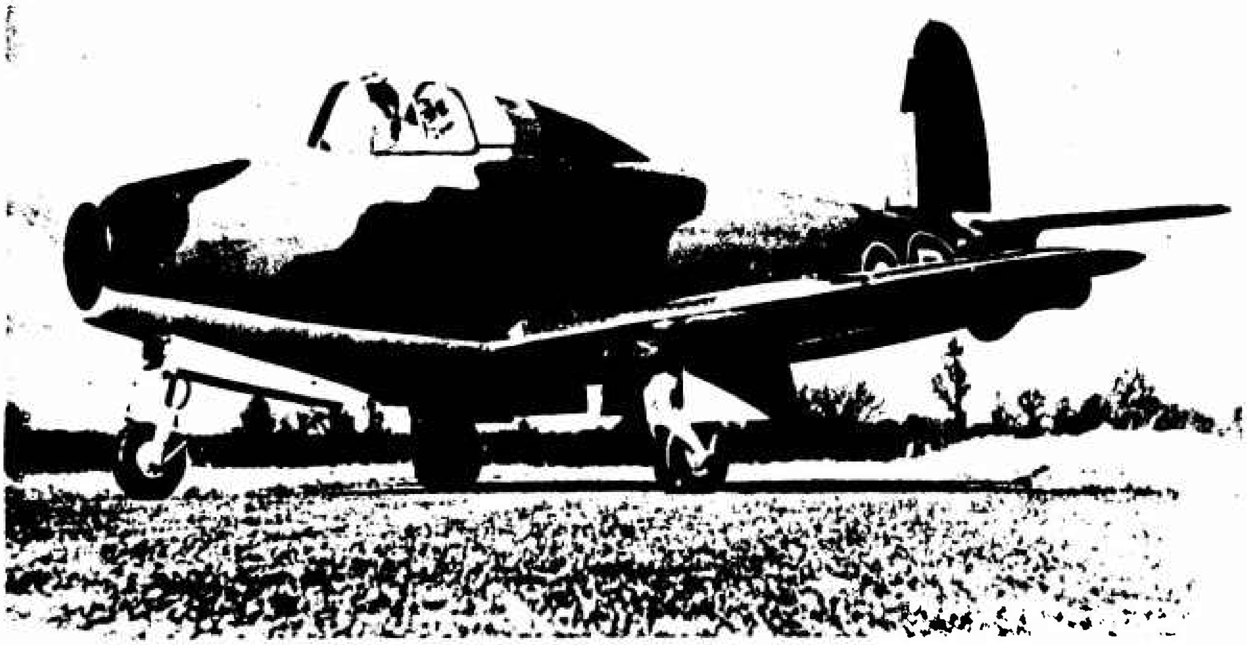
The last of the piston engined fighters, introduced at the end of World War II, were Naval aircraft such as the Sea Fury (single engine) and Sea Hornet (twin engined). These aircraft typify the classical layout of propeller driven fighters. With the advent of the jet engine, aircraft designers were forced to rethink the subject of engine installation and, in the first decade of the jet fighter, a number of solutions were tried. The first British jet to fly, the Gloster/Whittle E 28/39 adopted a layout which has since itself become a classic with undisturbed air taken in ahead of the airframe and the jet expelled clear of all structure at the rear. This



Sea Hornet

Sea Fury

Fig.1



Gloster/Whittle E 28/39

Fig.2

was followed by a number of attempts to redu e the volume of the installation, by shortening its length. Three examples are shown, the Vampire and the Sea Hawk adopted bifurcated intakes in order to shorten the inlet duct length, while the Meteor adopted two engines thus keeping both the intake and the jet pipes short. The Vampire used a twin boom fuselage tc achieve minimum jet pipe length (and maximum engine accessibility) while the Sea Hawk used an equally unconventional approach to keep the rear fuselage free of jet pipe, that is the use for the first time of a bifurcated jet with the flow split at the turbine rear face. With the Hunter, and later the Gnat, a 'conventional' layout was adopted with the pilot, radar ranging and guns taking precedence in the nose of the aircraft and relatively short bifurcated intakes supplying the engines which exhaust through fairly long jet pipes. The 1121, designed in the mid fifties, employed a single under-fuselage double shock intake with the engine moved rearwardly so that a relatively short reheat jet pipe exhausted behind the tail surfaces.

Other designers have shortened the jet pipe(s) by exhausting beneath the rear fuselage. This has been popular with twin engined types such as F89, F101, F4 and Jaguar. Later aircraft, such as F111, F14, and the Russian 'Foxbat', have reverted to the classic solution of exhausting aft of the empennage in the interests of obtaining low drag reheat-off and of eliminating heating and acoustic effects reheat-on.

Before passing to the detail engineering of powerplant installation I must mention an example of the extremely interesting features of VTOL powerplant installation. Figs 2-8 show the derivation of the powerplant layout in the P1127 family of aircraft, of which the present member, the Harrier, is now in service with the Royal Air Force. The top diagram shows that the first proposals, in 1957, had rotatable nozzles on the fan flow only, with a vertical force from the gas generator provided as a result of the angle of the powerplant relative to the ground in landing and take-off attitude. The front nozzles were to be rotated past the vertical to cancel horizontal forces. These layouts were, of course, wasteful and the proposal to bifurcate and rotate the hot efflux owed something to the earlier Sea Hawk experience. At this time also the 'bent pipe' exhausts were replaced with the now familiar compact and efficient vaned turning nozzles. The third diagram shows the resulting layout. At this point the separate intakes for fan and gas generator flows had been combined to feed a new design of fan and, on urging from the aircraft designer, the two spools were counter-rotated to eliminate gyroscopic coupling from the aircraft motions in V/STOL flight. Also the fan casing, which carried the engine fuel system and engine driven ancillaries, was rotated through 180° to bring these units to the top of the engine in line with the wing box, thus permitting a useful reduction in fuselage size. To this day the throttle linkage of the Harrier crosses over from the left hand side of the cockpit to the right hand

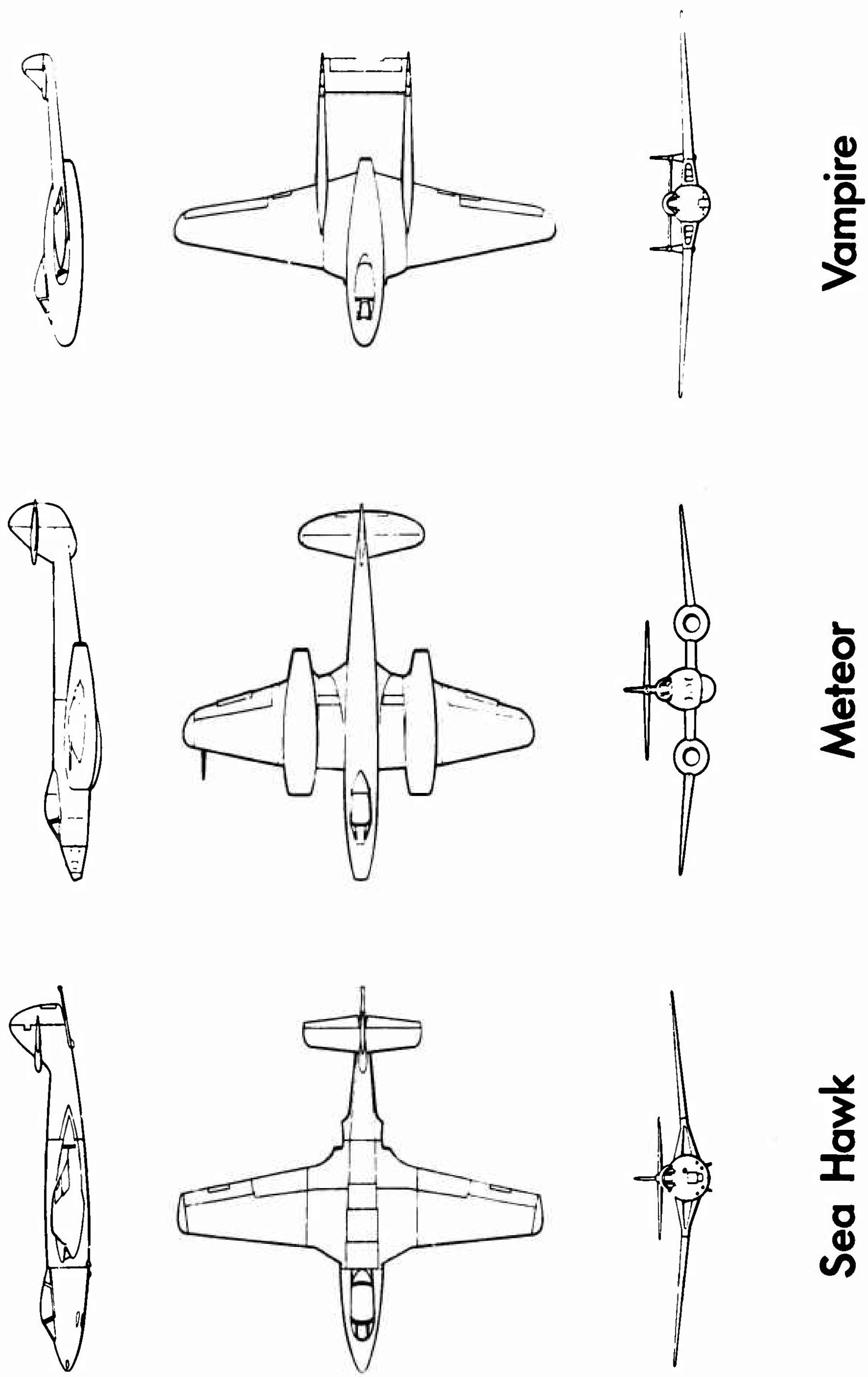


Fig.3

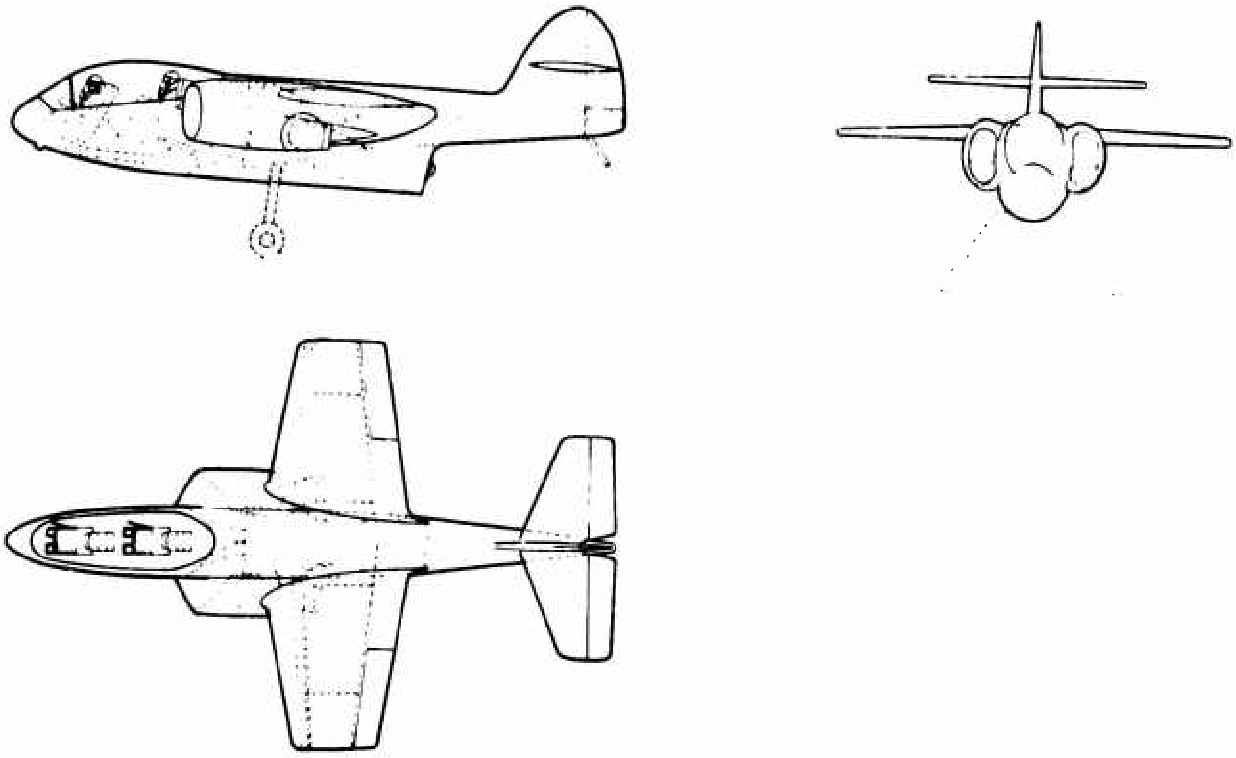


Fig.4 **P.1127** **July 1957**

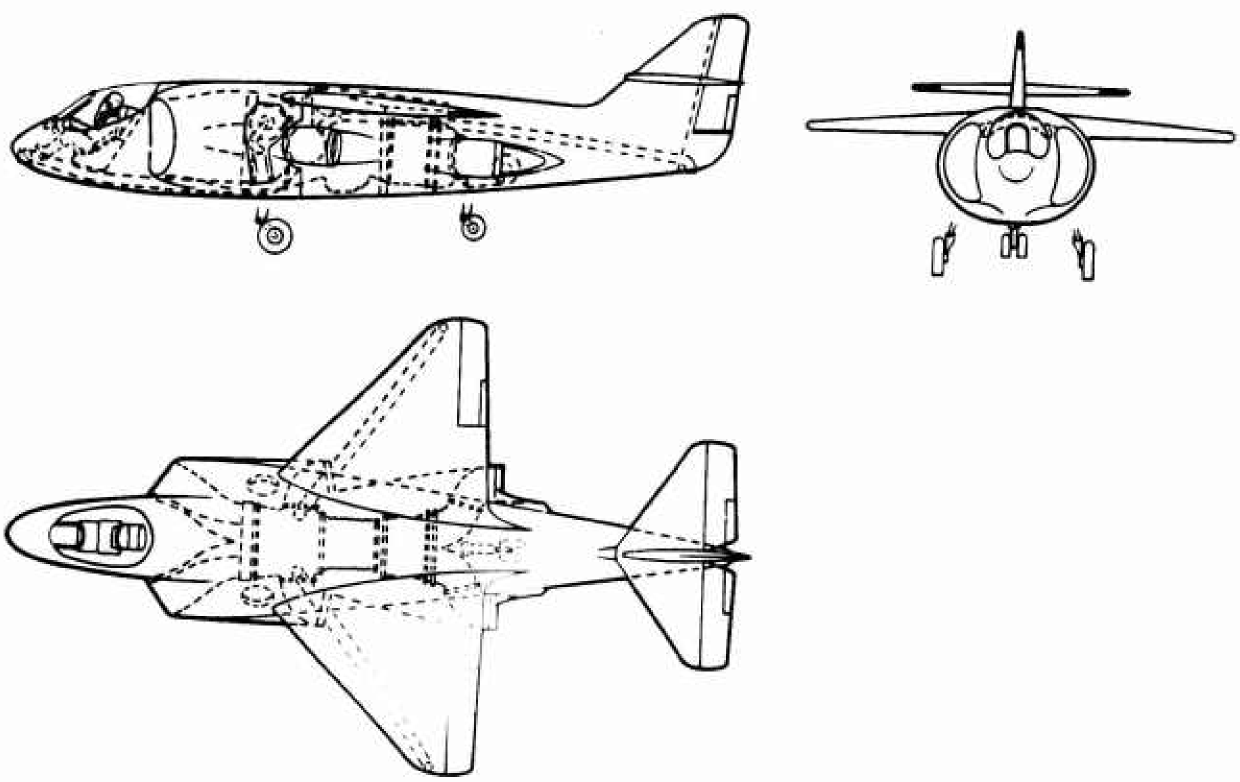


Fig.5 **P1127** **August 1957**

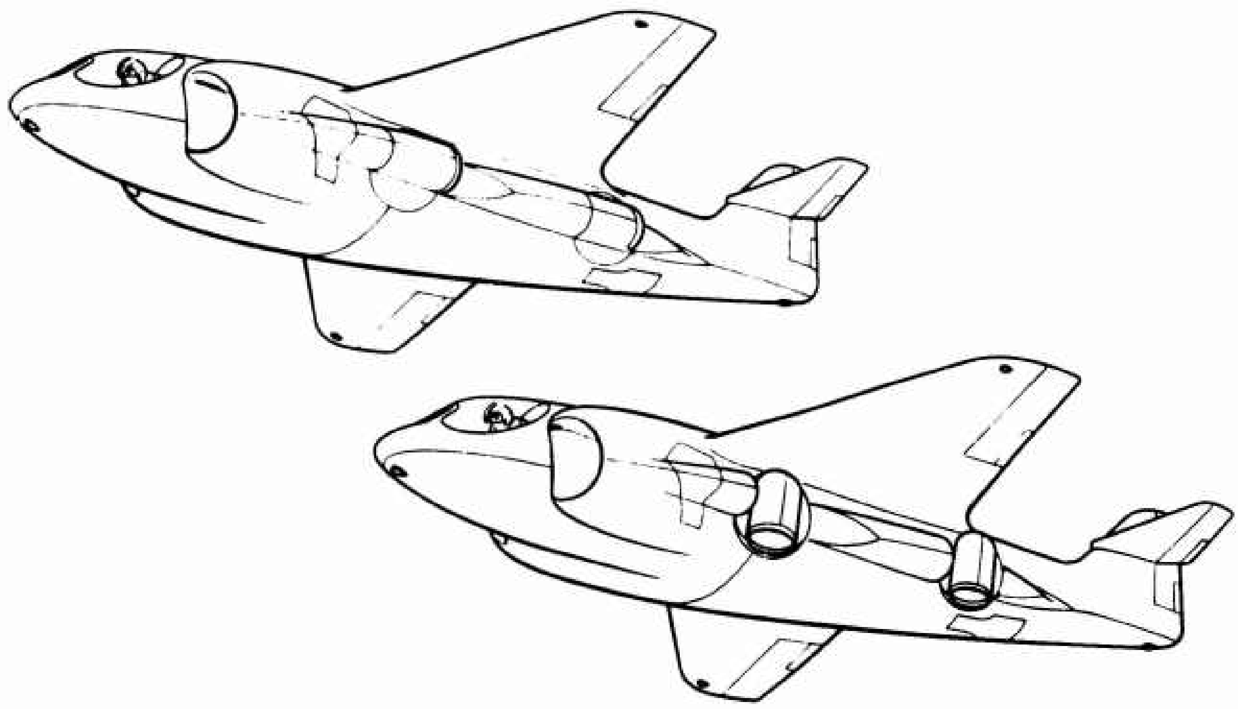


Fig.6 P.1127 August 1957

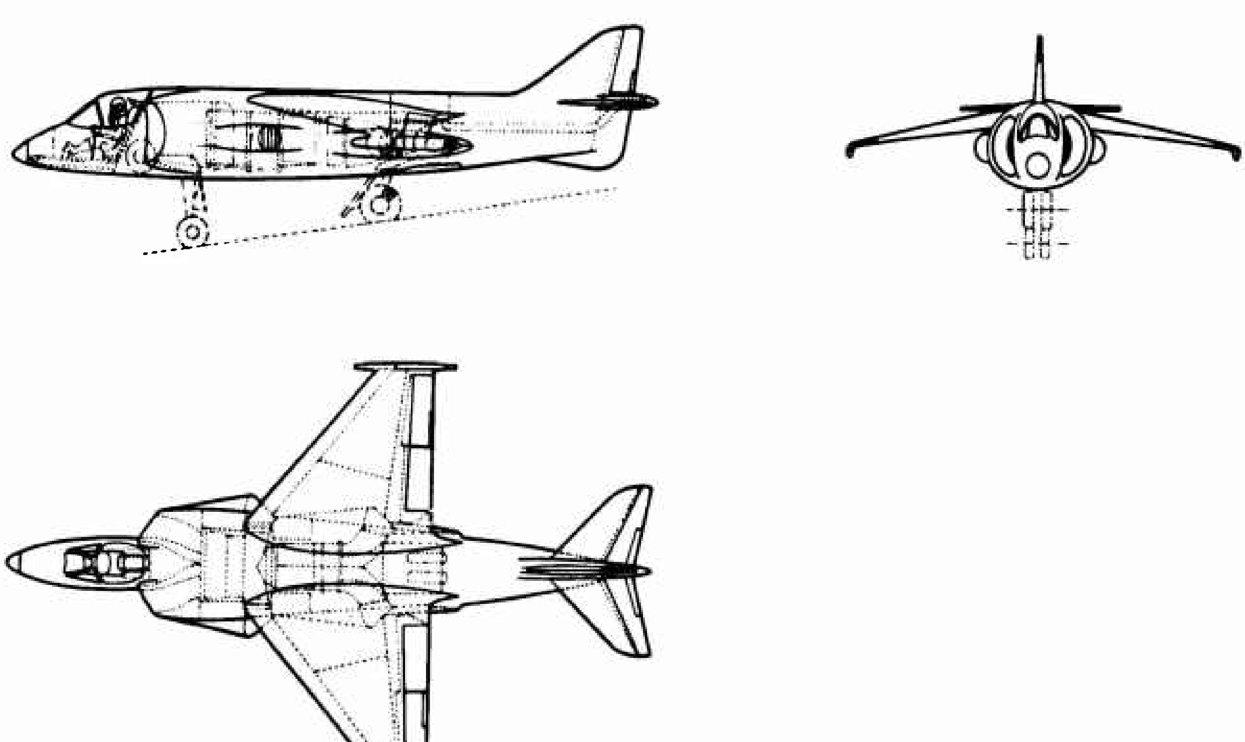


Fig.7 P.1127 March 1958

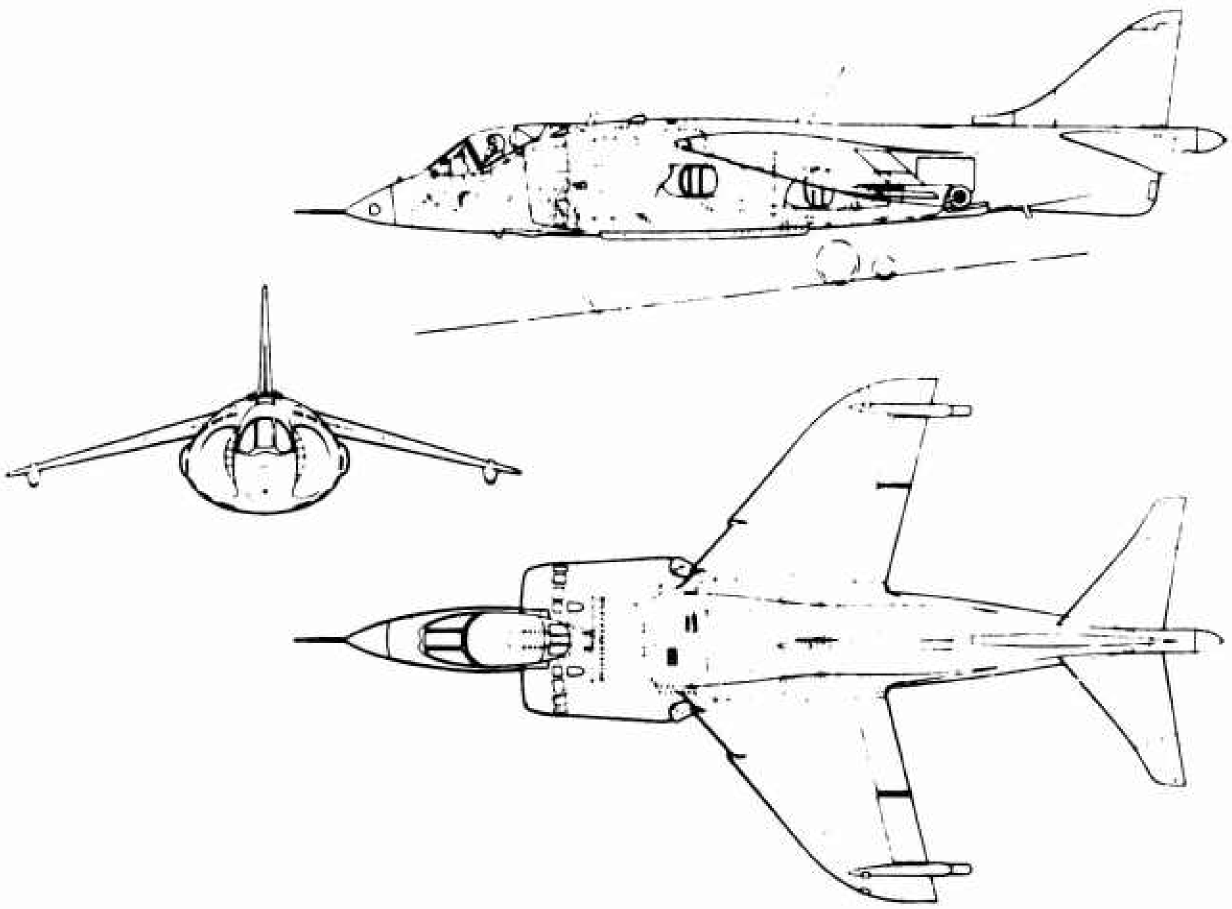


Fig.8 **Harrier GR1**

side of the engine, as a result of this early decision. The balance of a V/STOL design is a first consideration and, on the P1127, this necessitated a very short intake because the total weight of the cockpit and nose undercarriage would have been moved forward by any lengthening of the duct. This has led us to the solution of difficult problems in the intake and cowl design, which I shall return to later. With minimum length bifurcated intake and essentially zero length exhaust, the Pegasus probably represents the most compact powerplant installation, for its thrust, ever achieved on an unreheated engine — I should be careful to restrict the comparison to fuselage installations as some podded installations may be competitive.

I can scarcely do justice to the lift-engine concept of VTOL within the length of this paper; the layouts proposed have been legion and much has already been written on the many new features which have been resolved. Notable is the automation of the start up and shut down phases and the vectored installation proposed by R-R, E.W.R. and Fairchild Hiller Republic and obtained by the rotation of retractable pairs of lift engines on either side of the fuselage.

To summarise then, the evolution of fighter aircraft as affected by the jet engine has involved a search for a low volume installation while retaining good intake conditions and acceptable exhaust/airframe interactions. The pilot, his armament and the associated search and aiming equipment have often taken pride of place in the aircraft nose so that bifurcated intakes are common on single engined aircraft. This near necessity has been made a virtue on many supersonic types by the use of the fuselage side to house the variable geometry arrangements associated with multiple shock intakes. Twin engined types have tended to instal the engines further forward in the airframe because a shorter intake duct is permissible in the absence of bifurcation, and, as a result, the reheated exhaust has sometimes been liberated beneath the rear fuselage in an attempt to reduce weight. In spite of the attractions of the podded layout — so common in commercial aviation — the associated performance penalties have largely precluded their use on fighter aircraft.

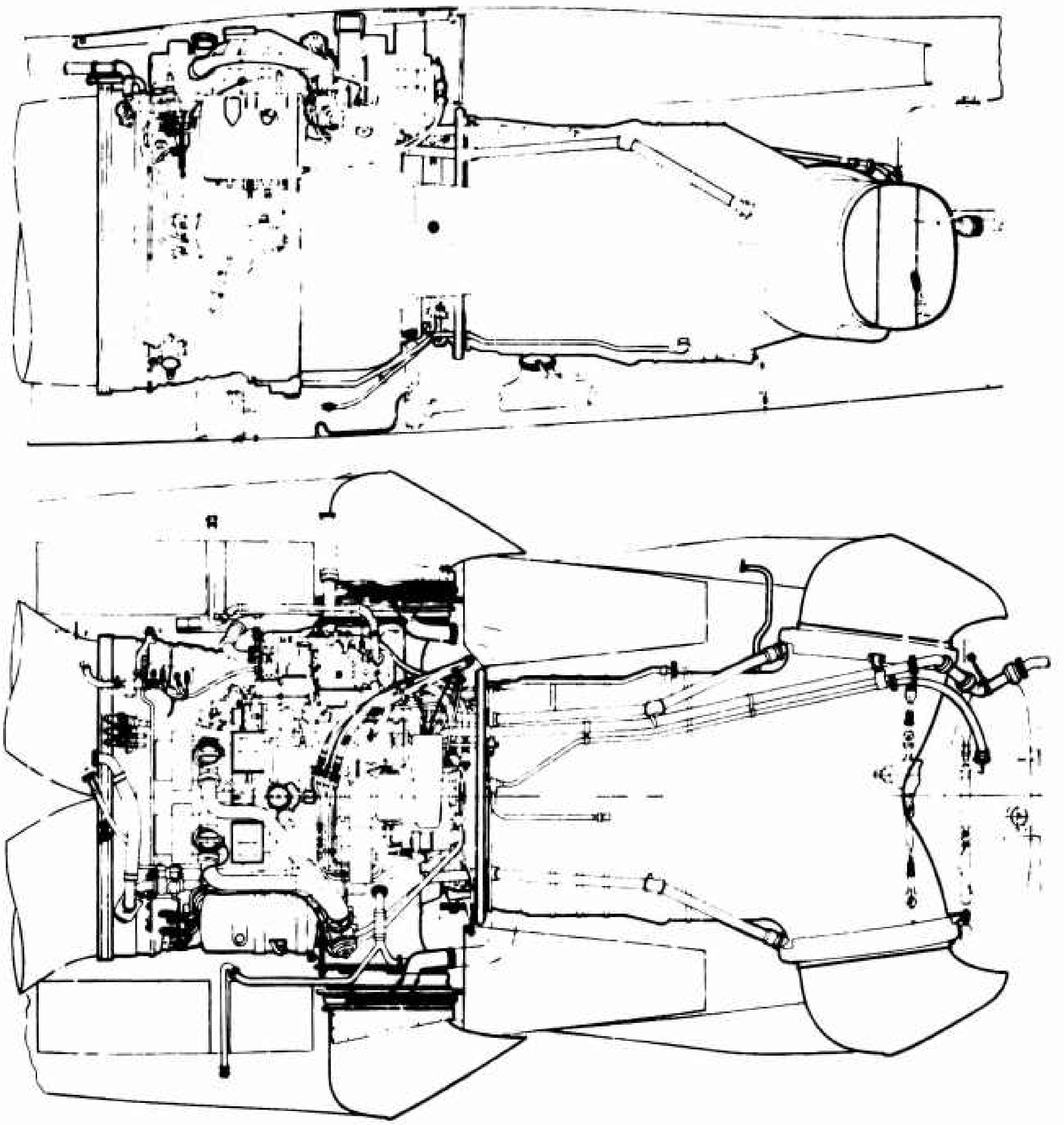


Fig.9 Harrier engine installation

4.0 THE ENGINE/AIRFRAME CONFLICT

The aircraft designer's requirements are simple. The powerplant should produce a large net thrust from a minimum size and weight with low fuel consumption. It should continue to do this for many hundreds of hours between overhauls. It should be rugged and very tolerant of the conditions of the air delivered to it.

In fact, the engine comes with a number of very undesirable by-products. It has a destructive exhaust, it produces excessive noise, infra-red radiation, vibration, smoke, heat (both internal and external to the airframe) and gyroscopic effects. It needs oil drains, fuel drains, it must be handled with care and mounted so as to escape the distortions experienced by the airframe. Its fuel control system is complex. Its intake flow is destabilising. It is expensive. If displeased with its environment it will surge, shed its blades or catch fire.

I must hasten to put the other side of the picture. The modern jet engine is a miracle of ingenuity. Compared with the piston engine it is free of vibration and it is extremely reliable. It is designed to breathe air but it is fed with rain, dust, grit, salt, rivet heads, grass, small stones, ice, birds and sometimes the exhaust from guns and rockets. The engine's theoretical performance is based on uniformity of air temperature and velocity across the intake face. With the advent of the VTOL fighter the engine must tolerate instead sudden changes of intake temperature or localised streaks of heated air. Also the aircraft designer has the

habit of demanding maximum thrust when the wing is hanging on in deep buffet with the intake at an angle which makes a mockery of the assumption of uniform flow. Aircraft designers have even been known to react with pained surprise when the engines quit in a developed spin — after all, if the airframe can recover from an incidence of 60^0 or more, surely the engine should continue to co-operate?

So, we are stuck with each other. Without his thrust we could do nothing. Without our drag he would have no job!

5.0 RESOLVING THE CONFLICT

I shall now discuss some of the installational aspects of the powerplant, with recommendations for both the airframe and engine designers.

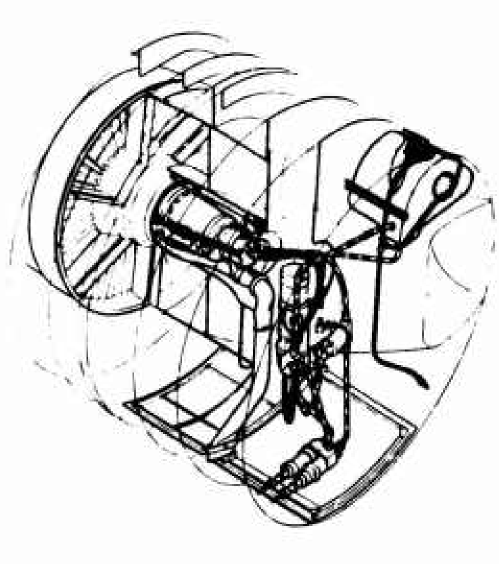
5.1 Starting

The early British jet engines were started electrically using ground supply. In the same period Germany was already using self-contained starting by means of small reciprocating engines mounted in the intake 'bullet'.

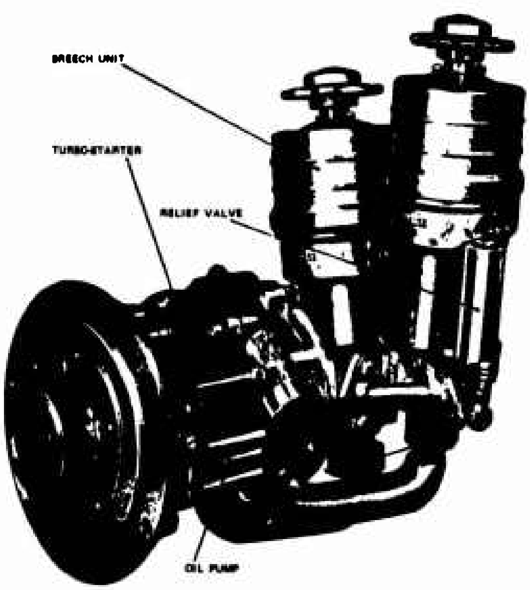
The early Hunters used cartridge starters and these in turn were replaced by a mono-fuel (iso-propyl-nitrate) starter with the same turbine impingement and clutch drive arrangement.

With the P1127 and its design emphasis on remote site operation, we reverted to cartridge starting. The cartridge was by now as big as the propellant charge of the gun armament of a destroyer and the cost of starting was high. The Harrier uses a gas turbine starter which burns a tiny proportion of the main fuel supply so that true independence of any special logistic support has at last been achieved. The gas turbine starter also doubles as a ground power supply A.P.U. and, while this function does complicate the installation, it is so clearly desirable for dispersed operation that I expect it to become the standard installation for future generations of military aircraft. The G.T.S. has proved to give very reliable starting, and with its direct mounting on, and drive-through, the main engine gearbox, testing during the Pegasus 6 development programme has been straightforward. It is worth noting at this point that the widespread adoption of twin shaft engines (pioneered by Rolls Royce, then Bristol Engines, on the Olympus) has greatly reduced the size of starter required since only the H.P. spool must be accelerated by direct mechanical drive. Presumably the new three shaft engines will continue this favourable trend.

I conclude that, following thirty years of operating jet aircraft, engine starting has at last reached a satisfactory stage of development. I believe that airborne relighting of the windmilling engine is also satisfactory on most modern installations.



Hunter
starter installation



Kestrel
starter installation

Fig.10

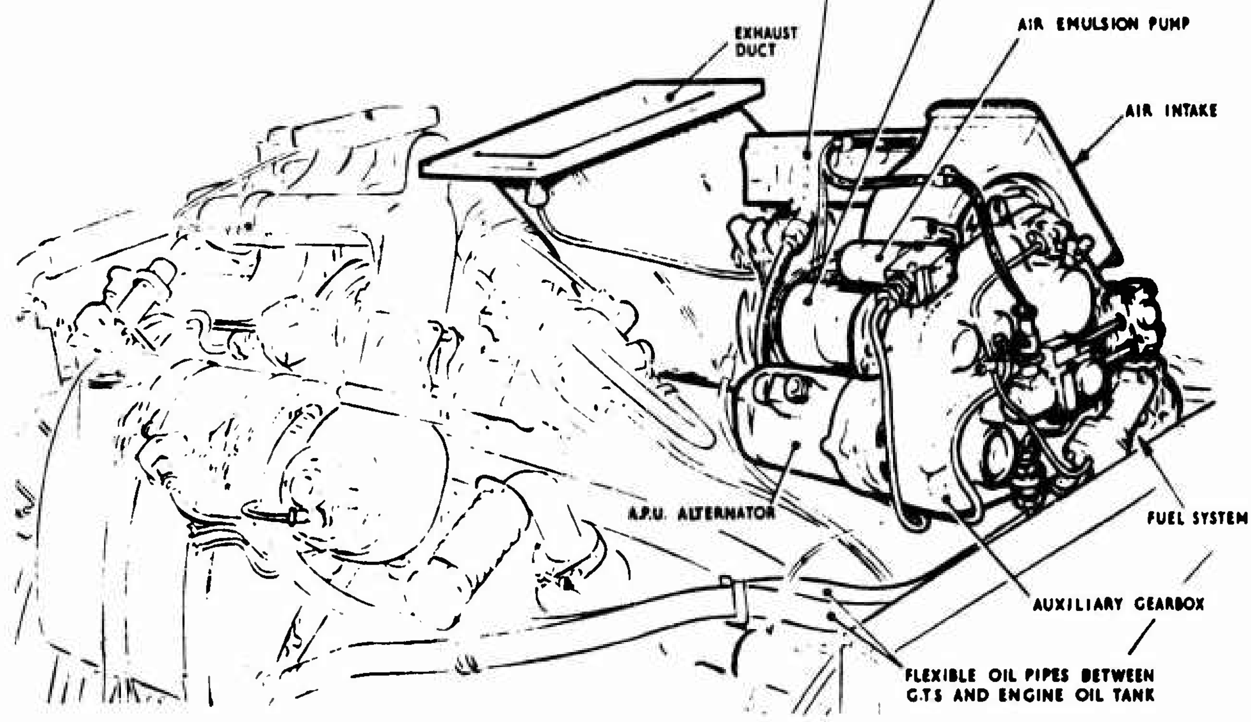


Fig.11 Harrier GTS/APU installation

5.2 Gyroscopic Coupling

The Pegasus engine pioneered the counter-rotation of the high pressure and low pressure spools, thus practically eliminating gyroscopic moments. This is particularly valuable in a V/STOL aircraft and it was implemented on the Pegasus, in spite of fears as to the satisfactory behaviour of the inter-shaft bearings, as a result of urging by the aircraft designer. A happy example of collaboration in the preliminary design stage, and one that has been so free of trouble that it is hard to understand why co-rotating twin-spool engines continue to appear.

5.3 Foreign Object Damage

In section 4.0 we listed some of these items. Firstly, rain must be regarded as a very normal operational hazard, but it has been known to chill the engine casings to the point at which blade rubbing can occur – usually in association with shaft bending under 'g'. This combination may, therefore, determine tip clearances and so affect compressor efficiency.

Dust and grit, in our experience, are unlikely to produce more than a small loss in the efficiency of compression due to roughening on the concave surfaces of the blading; but one hears from Vietnam of a fine grit which can rapidly modify the aerofoil shape of the small blades typically used in helicopter engines. This has led to the use of vortex air filters with considerable loss of performance. Such a solution would be impractical for the much larger mass flows of fighter type aircraft. We therefore look to the engine designer to come up with blade materials or coatings to stand up to this wear.

The problem of the prolonged ingestion of salt-laden air in shipboard operation, or even at coastal stations, militates strongly against the use of magnesium in both engine and airframe, except perhaps in the case of oil washed components or where inspection and replacement are easily accomplished.

The avoidance of rivet heads etc. passing through an engine is clearly the responsibility of the airframe designer, who must examine all fastenings in the intake ducts themselves and over the whole airframe area ahead of the intakes. The seriousness of such foreign objects has been much reduced with the advent of the by-pass engine with its relatively large and robust fan stages. It may well be possible to blend out the leading edge damage which results, just as was commonly done in the propeller era. The use of titanium blading in place of aluminium alloy has also improved engine tolerance to this type of damage.

Grass presents a new hazard, in the sense that only recently has the jet engined fighter ventured back onto the sort of grass fields used by the last of its piston engined forebears. This must become a greater cause of concern as the new military generation seeks to escape from the concrete base concept. Experience with the Kestrel and Harrier shows that uncut grass of

moderate height can be tolerated. However, if the grass is cut it must also be collected or quite large quantities may pass through the engine. This has not so far produced a dangerous situation, but a loss of engine performance may result. Under these conditions air tappings from the engine must either be filtered – and the filters must be easily accessible – or the piping must be clear of obstructions which might lead to clogging. An example of clear piping is the reaction control system of V/STOL aircraft, it has been noted many times during the development of the 1127 family that the reaction controls blow 'smoke' as the most obvious external evidence of heavy grass ingestion. The new biological (enzyme) cleaners show promise as a means of engine washing to remove grass contamination.

The ingestion of ice from the (unheated) intake lips of fighter aircraft has not, in our experience, been a cause of concern. The Pegasus engine pioneered the use of an overhung fan (now standard practice on the new high by-pass engines) so that bearing support struts ahead of the low pressure compressor were not required; from the Pegasus 3 onwards the inlet guide vanes were also deleted (again now standard practice) so that the first stage rotates. This has proved to provide complete protection against engine icing, the blades exhibiting good ice-shedding characteristics with no significant build up.

The rotating first stage of blading has also contributed to the ability of this quite remarkable engine to swallow the 600 kt. one pound bird which is constantly deemed to threaten low flying British military aircraft. Where static inlet guide vanes are featured, bird impact may force these back into the first rotating stage with disastrous results. Here the bifurcated intake gains some points, the cockpit providing considerable protection to the engine so that only the blade tips are exposed to the risk of direct impact. The problem is then transferred to the airframe designer who must build intake lips and the outside of the 'S' bend (Fig.12) such that the bird impact will not liberate pieces of metal of sufficient size to disable the engine.

Intake duct damage by bird impact test

Fig.12

Finally we come to gases ingested following the discharge of aircraft armaments. After overcoming considerable development difficulties in the early days of the Hunter due to unburnt gun gases causing rich extinction of the engine at altitude, we have managed to steer clear of this problem ever since. The Gnat fighter provides a fine example of just how intimate a 30 mm gun and an intake can get without producing flame-outs. Fig.13 shows that this has been achieved with the use of suitable gun gas deflectors. The Harrier has fired the total contents of six Sneb-Matra rocket pods *simultaneously* without suffering any adverse engine reaction. We may conclude that these problems rightly belong to the airframe designer and that, should he fall down on the job, then 'fuel dipping' as used on the Hunter may come to his aid.

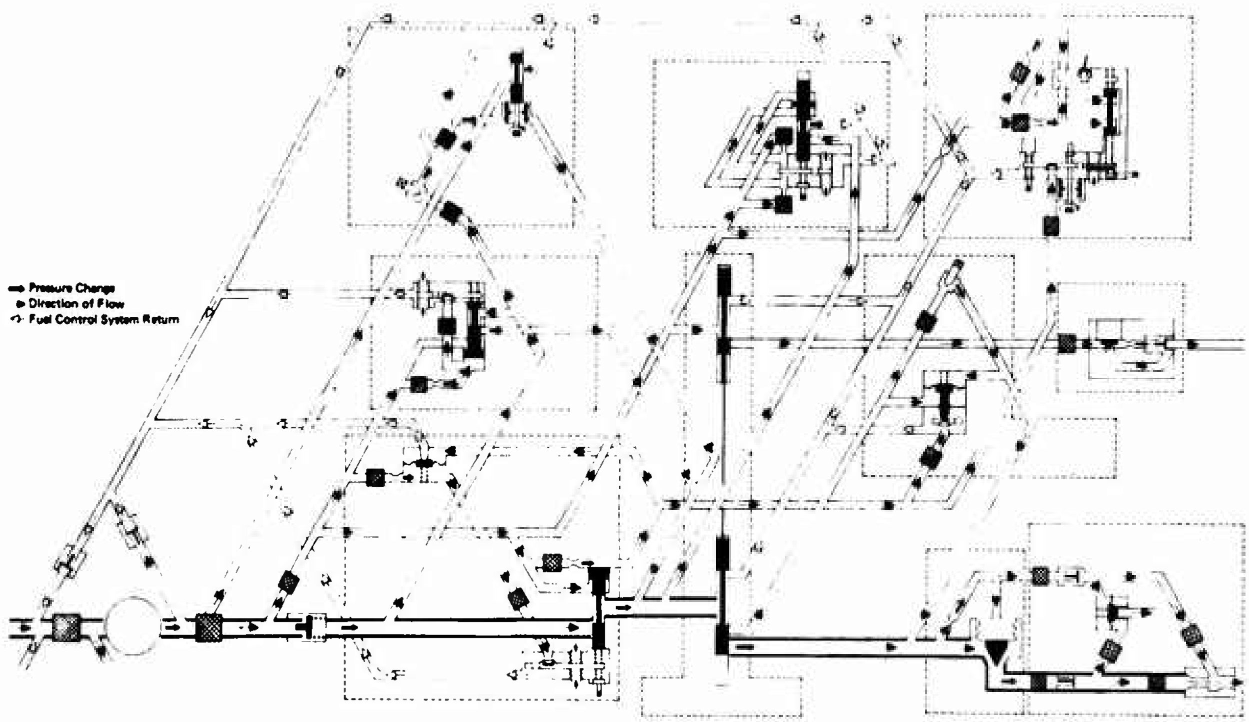
5.4 Fuel Systems

This interface between the engine and airframe is simple — it is the responsibility of the airframe designer to present air-free fuel within a designated pressure range and at a flow rate up to the maximum which the engine can consume at the inlet to the engine driven pump. The simplicity of this interface belies the complexity of the fuel system, both upstream and downstream.

On the airframe side it is necessary to pressurise the fuel system to control boiling at altitude and in order to transfer fuel from external tanks. The fuel remaining must be gauged and the flow rate measured with automatic compensation for change in density between different fuels; the fuel will usually be drawn sequentially from multiple tanks and metered so that the fuel load remains in balance both fore and aft and laterally. Arrangements must often be made for gravity and pressure refuelling on the ground and for in-flight refuelling. With fuel on board, a natural requirement arises for the aircraft to be defuelled on the ground and for the whole or part of the fuel to be jettisonable in flight. The pressurisation and venting system must cope with the maximum rate of change of altitude which the flight envelope will permit, and a trap system must be provided to allow a sufficient period of inverted flight at any fuel state. Assuming that these, and many other, problems have been overcome by the airframe designer, we may look downstream at the engine fuel system.

A typical British engine fuel system at the end of the first decade of the operational jet used twin variable stroke HP pumps with a servo-control system which received signals from throttle position, barometric (intake) pressure and an acceleration control unit. In addition, a mechanical governor limited over-speeding of the engine. In the event of mal-function of the servo-system causing loss of power, a pilot-operated switch would isolate one of the HP pumps which then delivered fuel at full stroke. With the addition of a J.P.T. control and fuel dipping for high altitude gun firing, this system remains in the Hunter today.

Fig.13 Gnat Fighter



Harrier Hydro-mechanical engine fuel system

Fig.14

With the advent of the two-spool engine and the special requirements of V/STOL operations, the hydro-mechanical fuel system has grown into an electro-hydro-mechanical system with a number of new features. The controlling parameters now include:—

- Throttle angle
- LP speed
- HP speed
- Fan output pressure
- HP output pressure
- J.P.T.
- Intake temperature
- Water injection flow
- Reaction control flow

In various combinations these parameters are fed to the:—

- Pressure drop regulator
- LP governor (including water injection reset)
- Acceleration control unit
- Pressure ratio limiter
- Air bleed reset
- Flow control pressure drop regulator
- Inlet guide blade control unit*

* *This unit may be automatic in operation and, although using engine fuel as its operating medium, it may not control the engine fuel flow directly.*

Such a system has now reached a satisfactory state of development and has met all requirements placed upon it. However, it is complex and bulky and time consuming in development. Perhaps the time has come to think again.

Why has the engine fuel system reached this level of complexity? (Fig.14). There seem to be two main reasons, the first is that the components of modern engines are designed to exploit their performance close to the limit, the second is that the ill effects of exceeding limitations may be hidden from the pilot (e.g. excessive turbine creep rate) or may be sudden and potentially disastrous (e.g. surge and run-down).

As each new limitation has been approached, therefore, a new appendage has appeared on the engine fuel system designed either to reduce the cockpit work load by limiting the number of gauges which the pilot must scan and interpret, or to protect him where the rate of scan required to anticipate trouble is beyond human capacity.

With the rapid advance of digital computation, it is now possible to make a comprehensive digital model of the engine behaviour with unfailing memory and a scanning rate many orders higher than that of the human brain. Such a device could incorporate multiple redundancy to ensure a high overall integrity or self-checking with fail-safe may offer an alternative.

In either case a final fall-back position comprising an old-fashioned mechanical linkage from the pilot's throttle to a fuel tap on the engine would allow the pilot to get home; if with rather unenterprising use of the potential engine power.

Digital control offers a wide range of new possibilities. For example, a tie-in with the aircraft inertial and air data systems, as a means of sensing pitch and sideslip angles, together with a knowledge of intake flow distortion characteristics could provide automatic protection against engine surge due to this cause. A cockpit gauge could indicate continuously to the pilot the percentage of maximum safe power which he is demanding. This indication would vary, at fixed throttle setting, during aircraft manoeuvre or other transient conditions. Auto-throttle control for blind approach or during the attack phase are other obvious by-products of this system.

Further in the future the functions of the engine digital fuel system may be taken over by a central digital computer which deals with the aero-dynamics of the vehicle, the electronics of the nav/attack system and the thermodynamics of the powerplant. At this stage the engine fuel system is no more than the fuel tap with standby mechanical control as envisaged above.

5.5 Intake Design

The satisfaction of the engine's demand for air has grown from no problem at all in the days of the centrifugal powerplant to perhaps the most difficult aerodynamic problem in the design of a V/STOL fighter aircraft with a high by-pass lift/cruise engine.

Essentially the requirements for high static efficiency conflict strongly with the requirements for the throttled cruise, particularly so if the cruise is at low level. In the latter case the external flow around the intake cowl may assume an importance equal to that of the internal duct flow. The engine designer may count himself fortunate to be able to concentrate primarily on the flow within the bounding streamlines of the capture area. In general, the conflict referred to above can be solved only by recourse to some form of variable geometry. On the Harrier we have sized the intake throat area for the high power, high altitude case. This, in combination with (a) a large highlight-to-throat area ratio of the intake lips (b) a generous secondary duct area exposed by blow-in doors in the external cowl (c) provision of doors to close off the boundary layer bleed (thus preventing the circulation of low energy reverse-flow in the V/STOL regime) (d) close attention to the shaping of the duct, particularly where this can improve the flow into the inner annulus and hence into the HP compressor of the engine; has resulted in a high static efficiency.

In the cruise the blow-in doors are shut and the BLB doors are open. In manoeuvring flight at either very high or very low incidences (and depending on RPM and speed) one or other of the BL doors may be shut under the influence of the pre-entry compression and the upper or lower inlet lip suction. The external cowl has been shaped to make the best use of the available cowl/highlight area ratio.

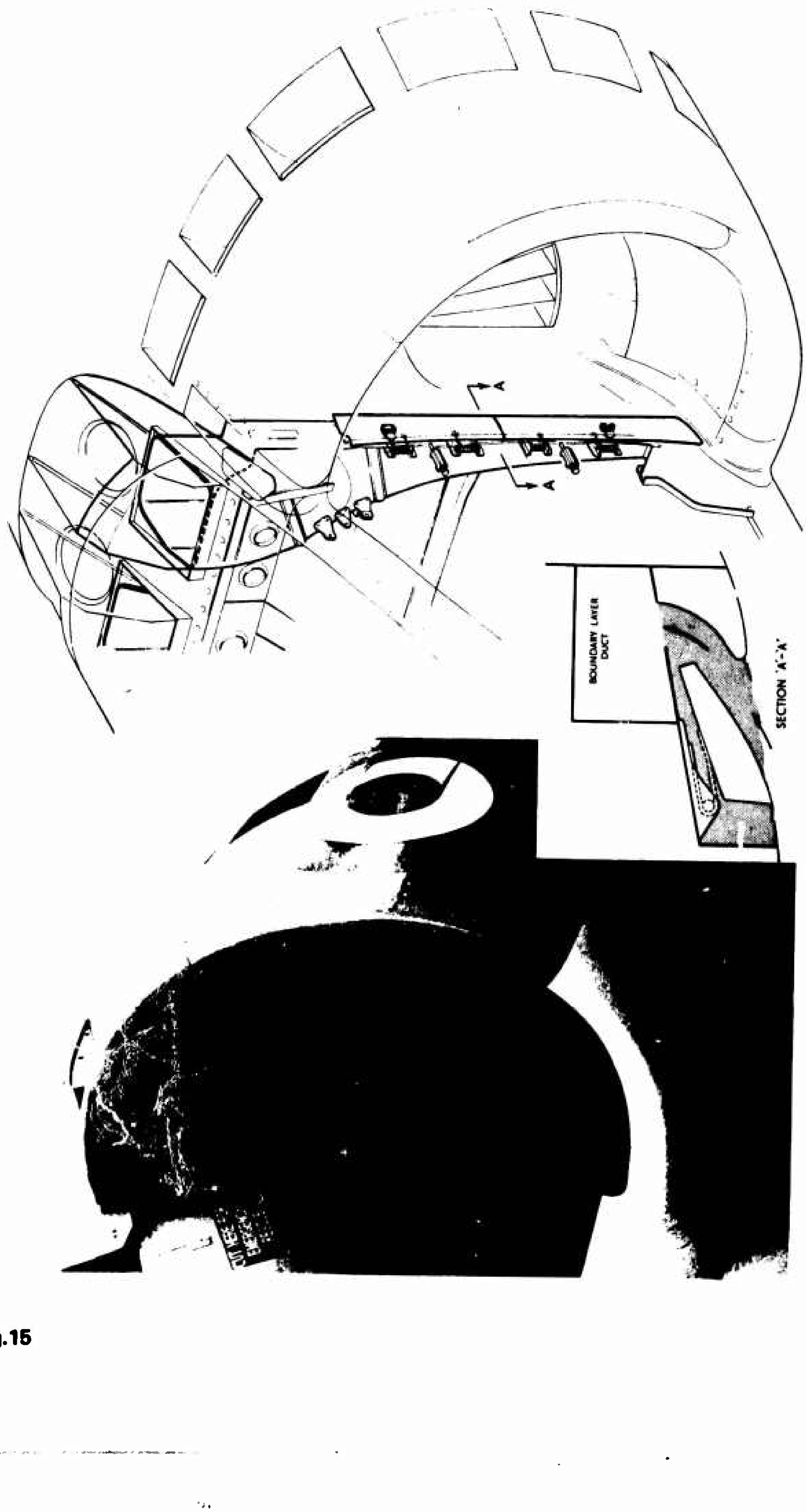
In the course of the development of the 1127 air intake, from prototype to Harrier, we have investigated turning vanes, vortex generators, distributed suction, B.L. blowing by HP or LP engine air, translating lips, hinging lips, inflatable lips and a number of other aerodynamic tricks. Fig.16 shows an alternative arrangement to that finally adopted on the Harrier. Neither has the powerplant designer been idle, although his activities have often been of a defensive nature – by which I mean that he must ensure that the engine performance, handling and mechanical integrity will tolerate the presence of the airframe.

I have dealt briefly with the intake of a particular aircraft during the development of which we learnt that the long fan blades and increasing stage loadings of modern high-bypass engines may be quite sensitive to the wide variety of velocity patterns produced by the air intake over the flight envelope of a fighter aircraft. The lesson for future preliminary design of similar aircraft is that intensive work on the intake should start at an early stage. If a new engine is to power the new aircraft then it may be desirable to run a model intake ahead of a model compressor in order to obtain early indications of the engine response.

To an airframe designer it seems that the problems of engine design – involving the interaction of n stages with x_n blades, on y spools, and with the blade boundary layer subject to centrifugal loading – are so daunting that perhaps we should mend our ways and use podded engines, programmed by the computer envisaged in the last section to point always directly into the relative wind! (Fig.18)

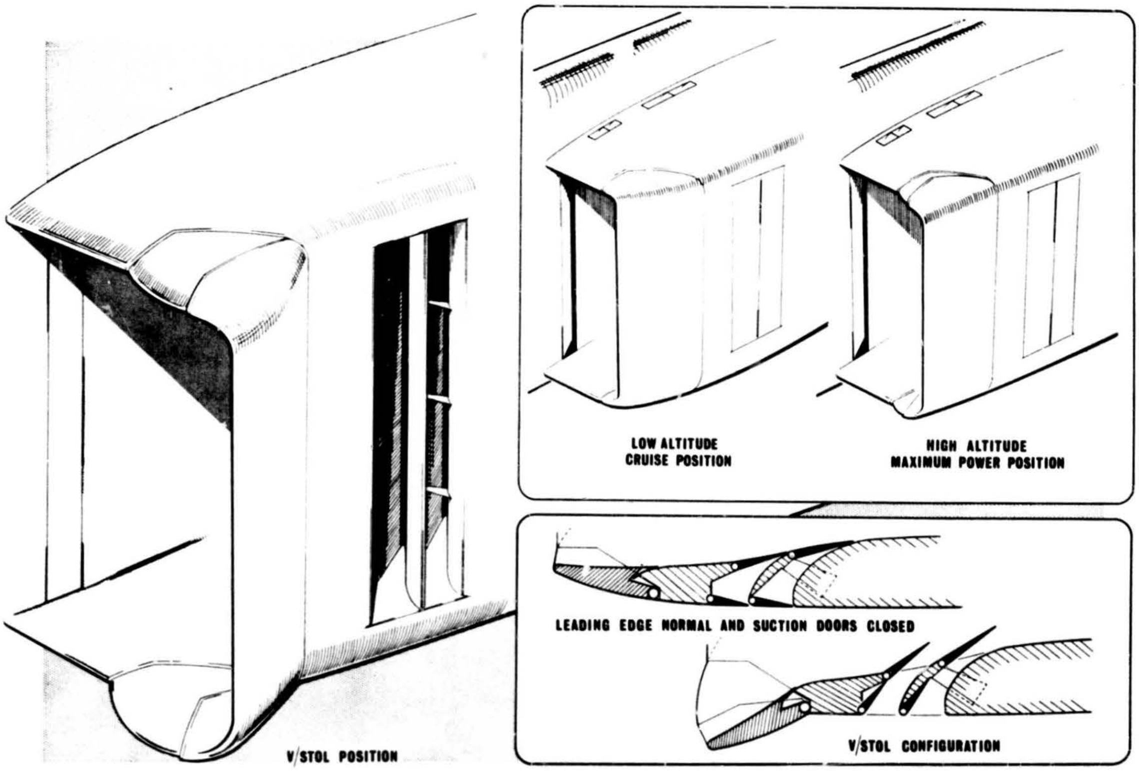
5.6 Exhaust Design

If the exhaust could be considered in isolation then there can be little doubt that it would be situated at the extreme rear of the airframe. As already noted in section 3.0, considerations of space and weight (or special function as in the case of vectored thrust) may result in the jet exit(s) lying beneath or in the flanks of the rear fuselage. In these positions problems of structure heating and acoustic fatigue arise and close attention must be paid to rear fuselage drag. These problems exist with the simplest type of exhaust and they are multiplied by the requirement for reheat with its large variable convergent nozzle.



Harrier intake features

Fig.15



Alternative V/Stol air intake configuration

Fig.16

Further complexities arise with high supersonic speed and with the demand for short landing ground runs. The first leads to convergent-divergent nozzles and the second to reverse thrust.

To the writer's knowledge no purely mechanical variable convergent-divergent nozzle has flown, but an approximation employing an induced secondary airflow (reheat-off) is in widespread use. The combination of this arrangement with thrust reversal is used for the first time on a production aircraft by the Saab Viggen. A demand for the reversal of re-heated thrust may well arise in the future.

Vectored thrust offers a simple means of complete reversal although this has not yet been implemented on the 1127 family. In general a demand for a high degree of reverse thrust will set new constraints on the aircraft configuration in order that trailing edges, external stores etc. are not exposed to excessive temperature or jet blast.

In summary the exhaust system, required to operate efficiently from full reverse, through the subsonic cruise and at high supersonic speed, will be a complex and difficult piece of high temperature aero-mechanical engineering. There is great scope for improvements in the state of the art and for collaboration between engine and airframe designers.

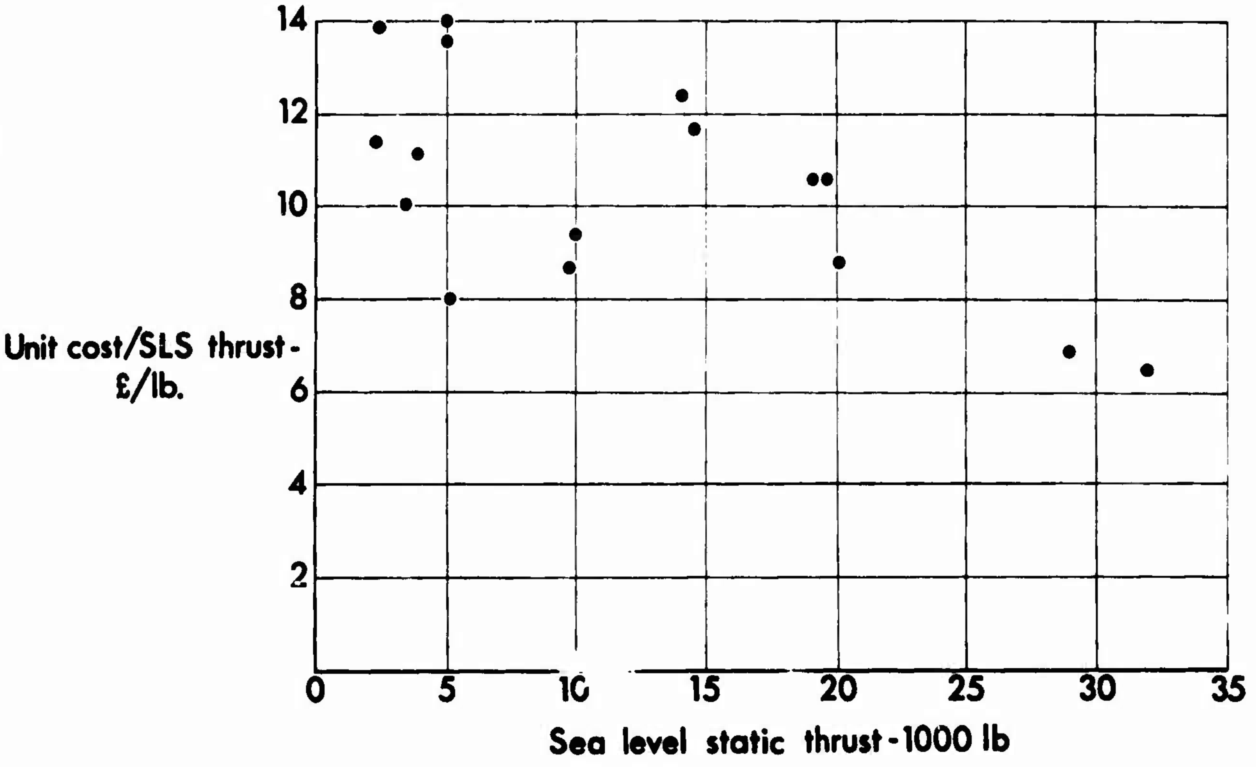
5.7 Powerplant Cost

Unit cost is increasingly a powerplant problem. The aircraft designer must think seriously before introducing a new engine for each new aircraft design, although the flexibility of the building-blocks of the jet engine make this approach technically attractive.

Fig.17 shows recently quoted unit specific costs for a number of European and American engines. The scatter is large as corrections for the length of production run and the degree of 'dressing' of the engines have not been made. However, it may be concluded that the penalty for adopting the latest powerplant will be to increase the cost by 50–100% as compared with an older and proven design.

The availability of engines in particular thrust brackets together with the costing methods adopted, may well sway the choice of aircraft layout as between a single or twin engined design.

The powerplant, the airframe and the electronic systems now contribute roughly equally to the unit cost of a modern strike fighter. We must, therefore, ask the engine designer to pay close attention to cost at the preliminary design stage.



The cost of thrust

Fig.17

6.0 CLOSING REMARKS

The speaker has learnt to admire the engine man with his complex problems – but does the blading always need to have such sharp leading edges even where the relative flow is subsonic? Must by-pass flows always pass through such aerodynamically dirty passages? Do any of the many variables in the engine design – multi spools, variable guide vanes including variable turbine stators etc. – hold out any possibility of varying thrust over the operating range while maintaining air mass flow relatively constant? Will the stiffness of carbon fibre blading see the end of the fan blade 'snubbers' which are in common use today? What happened to the super fuels so much in vogue a few years ago – are there to be no new developments on this front? Will there ever be a better way to produce thrust than slapping the air with a series of blades and then setting fire to it?

My theme has been that the preliminary design of both powerplant and airframe is heavily circumscribed by practical considerations. The mutual interaction of airframe and engine is usually unfavourable† and only by close collaboration by the two design teams will a successful compromise be reached. The search for the ideal thermodynamic cycle for a given specification condition (or series of conditions) is only the initial scratch on the surface of preliminary design.

Finally, I would like to express my gratitude for the stimulating co-operation which it has been my good fortune to enjoy with the powerplant manufacturers – now *the* manufacturer – in the United Kingdom. Their problems are often of our making.

In this short paper I have left many subjects untouched but I hope that the contents may stimulate discussion. The opinions are, of course, my own and do not necessarily reflect those of Hawker Siddeley Aviation, whose permission to present this paper is gratefully acknowledged.

† *Except that, by the presence of an engine, a powered aircraft is produced which would otherwise have been a glider!*

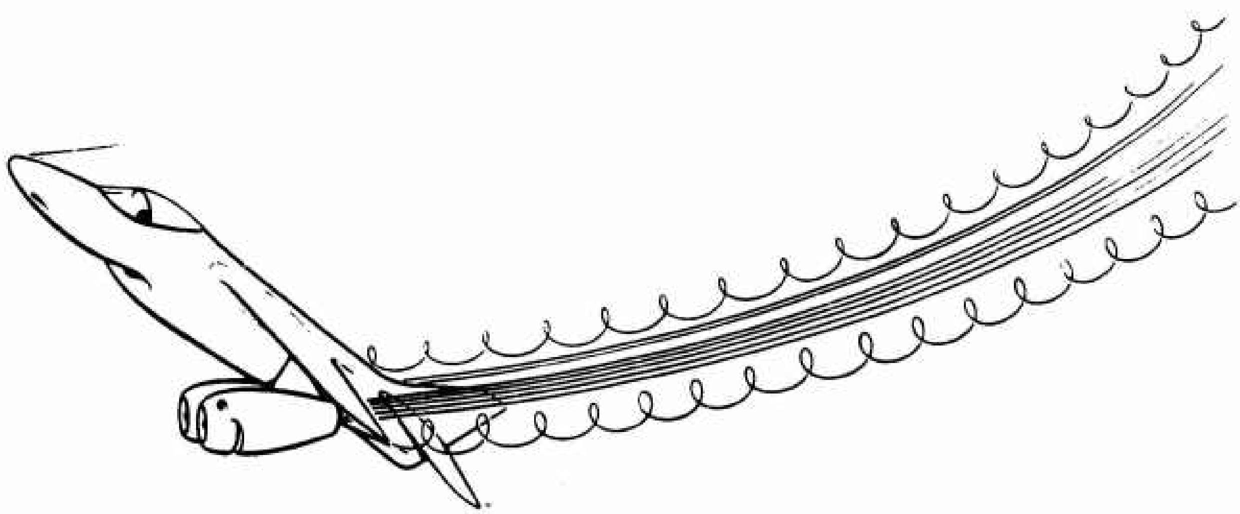


Fig. 18

"...programmed by the computer to point always directly into the relative wind"

THE DESIGN PROCESS

by

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General Electric Company
Cincinnati, Ohio, U.S.A.

THE DESIGN PROCESS

by
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ABSTRACT

Current trends in military combat aircraft are clearly towards increased emphasis on multi-mission capability. This influences engine design by requiring multiple thrust sizing points throughout the flight envelope and good cruise performance over a wide range of flight conditions and power settings. Engine performance, dimensions and weight cannot be treated as isolated elements. The engine cycle cannot be selected on the basis of uninstalled performance with subsequent efforts made to minimize the penalties associated with the inlet, exhaust and other installation factors. The engine manufacturer must expand his area of interest and treat the engine as an integral part of the overall system throughout the design effort. The initiation of engine development programs independent of the weapon system will no longer provide the required results.

The system design approach starts with the identification of a General System Concept and continues through Concept Formulation, Contract Definition, and the actual Development Program. While engine development is recognized as requiring an extensive time period, an engine for an advanced weapon system also must include as many as five years in the definition period prior to contracting for engine development. This very important formulative phase is used to identify critical state-of-art advancements needed to provide mission performance significantly beyond existing capability, and to initiate development efforts to verify the feasibility of these improvements prior to program commitment. Flexibility to cope with changing needs must be provided throughout this time period as well as maintaining a proper balance between performance, weight, and technology versus cost, schedule and operational requirements.

—*—*—*—*—*—*—*—*—*—*—*—

Introduction

Current trends in military fighter aircraft are clearly toward increased emphasis on multi-mission capability. This type aircraft, in stages of either conceptual design or development are highly complex weapon systems made up of equally complex and advanced sub-systems.

The diverse capabilities of such aircraft require an unusually wide range of propulsion system thrust levels to provide excess energy needed for combat maneuvers in addition to low fuel consumption at reduced thrust levels for cruise operation. The flight envelope is broad, encompassing the loiter, transonic combat and supersonic intercept demands of this aircraft. Figure 1 illustrates the important design areas of the propulsion system.

As a consequence of this broad capability, the design of the propulsion system must be highly integrated into the total system and requires design processes that consider all aspects necessary to achieve maximum weapon system performance.

Design Processes

The design processes which must be employed in the conceptual and physical development of an advanced propulsion system are characteristized by a series of key decisions at strategic points of time that are important to the development to both the aircraft and the engine. These key decisions are listed on Figure 2. Ultimate program success is dependent upon selecting the proper courses of action to be taken at these decision points. A brief discussion of some of the considerations at these key program decisions is appropriate at this point.

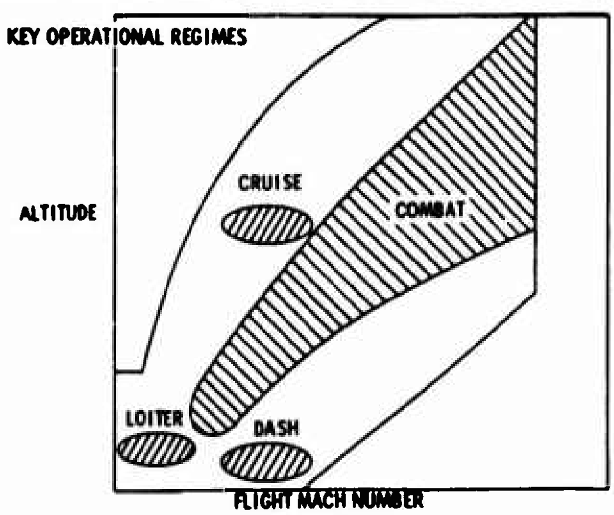

Figure 1

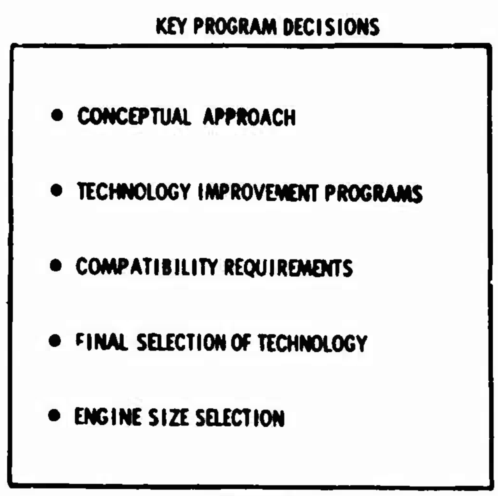

Figure 2

Conceptual Approach

The system design starts with the identification of a conceptual vehicle aimed at meeting a general set of operational requirements. These usually define several thrust sizing points in addition to two or three mission profiles. The thrust matching and mission performance relative to these operational requirements define the basic engine characteristics and determine the engine type which most efficiently meets the requirements, be it turbojet, turbofan or some more exotic cycle.

Tradeoff studies are performed on an overall propulsion system basis with proper consideration of inlet matching, afterbody/exhaust nozzle interactions, environmental conditions, bleed and power extraction, maintainability, reliability and vulnerability requirements; all of these in addition to the specific performance requirements. In support of these studies, the engine manufacturer furnishes parametric data which describe the effects of design parameters such as bypass ratio, turbine temperature and pressure ratio on overall performance. Figures 3, 4, and 5 show examples of the type of data furnished for augmented turbofan engines. With these data, parametric mission analyses are performed with a variety of aircraft configurations to yield an initial definition of the engine cycle, airflow size, thrust level, aircraft size and drag. These analyses form the basis for the next key decision.

PARAMETRIC DATA

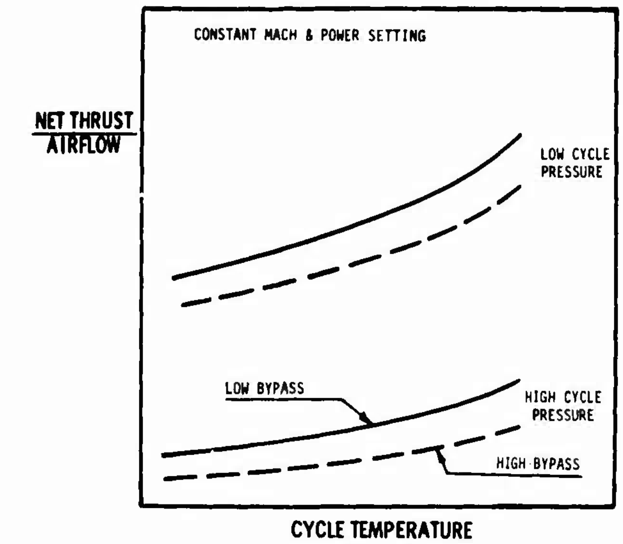

Figure 3

PARAMETRIC DATA

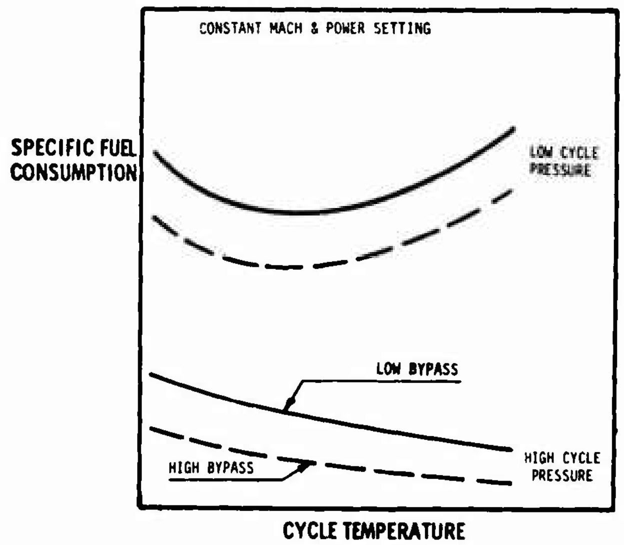

Figure 4

PARAMETRIC DATA

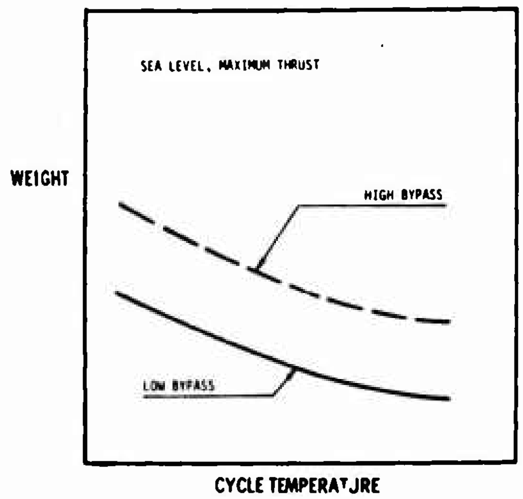

Figure 5

Technology Improvement Programs

Engine development is generally recognized as requiring a longer period of time than that required for the aircraft. Figure 6 shows a typical development cycle for an advanced engine. The identification of critical, but achieveable, state-of-art advancements needed to meet system requirements is very important. It is equally important that development programs to achieve these advancements be initiated with flexibility provided to cope with changing needs.

The decision to implement these development programs must be based upon their value to or impact on the propulsion system. The study results, from the conceptual approach, will allow a decisio.. to be made. Prior to implementation of the programs, firm planning through completion of the program must be made. This includes the time to execute various phases and the cost. Comparison of these estimates, relative to available resources, must be made such that the greatest improvements can be achieved with minimum cost. Decisions with these considerations in mind have major impact on the ultimate engine program.

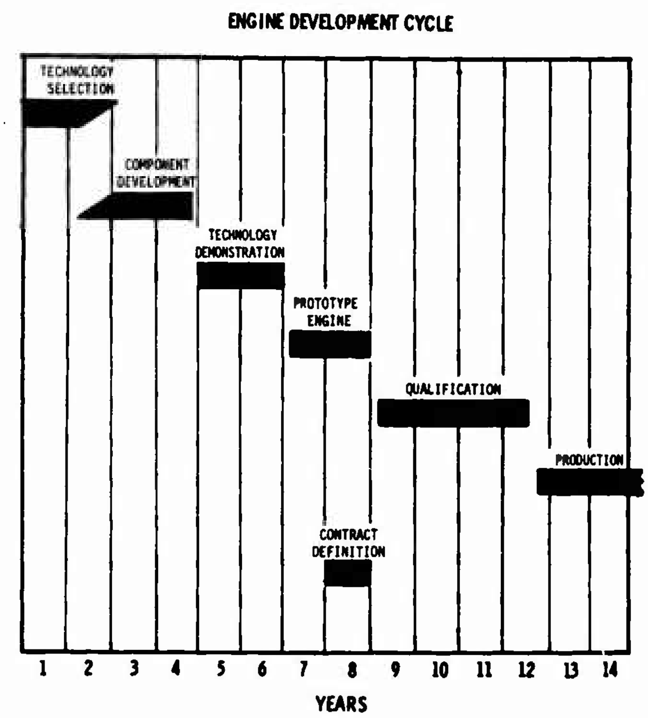

Figure 6

An example of a comprehensive Technology Improvement Program is the GE1 Advanced Engine Program, begun in 1962. It was conceived as the progenitor of a stable of engines that would meet a variety of propulsion system requirements. Through judicious choices of technology advances, different engine types were evolved and successfully demonstrated; a medium thrust/high pressure ratio turbojet (augmented and non-augmented), a high bypass ratio subsonic turbofan; medium bypass ratio augmented turbofans. These technology advancements include: significant increase in turbine inlet temperature; higher loading in the turbomachinery; development of new mechanical designs and manufacturing techniques; material improvements. The progression of this program is illustrated in Figure 7.

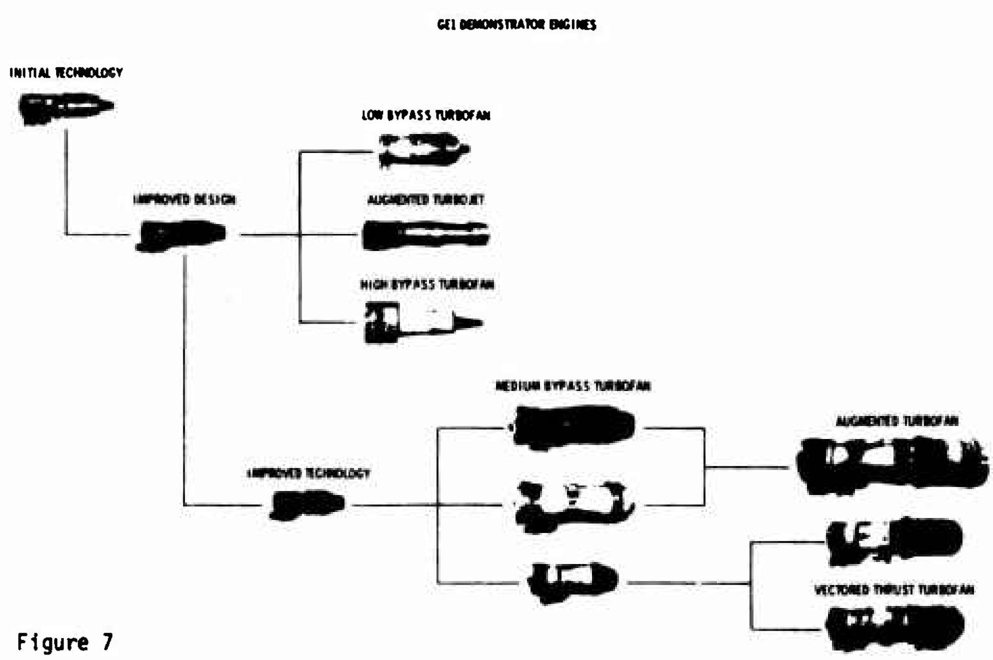

Figure 7

Compatibility Requirements

The integrated system approach continues from the conceptual phase into the design phase and throughout the development phase to insure a compatible engine/aircraft system for operational service. In addition to the normal mechanical interfaces, compatibility can be grouped into basically two areas:

- Thermodynamic compatibility
- Aerodynamic compatibility

Each of these will be discussed briefly.

The combined requirement of high performance and mission flexibility (or thermodynamic compatibility) necessitates a new approach to integration. The engine, inlet and exhaust nozzle must be thought of as a system and are key components in aircraft configuration and performance. No longer can an engine cycle designer ignore factors of aircraft performance. Likewise, the aircraft designer must consider engine performance characteristics and the possible tradeoffs. The traditional process of selecting an engine cycle based on uninstalled performance and then applying effort to minimize the installation penalties is obsolete.

One aspect of installed engine performance is that of inlet/engine airflow matching. Since inlet capture area is normally sized at the maximum Mach flight condition, the inlet flow will be in excess of engine requirements for all lower speeds and at low altitudes. Combining engine and inlet characteristics for maximum installed thrust or minimum spillage results in the general characteristics shown in Figure 8. These losses are significant. However, greater losses are incurred during subsonic cruise operation as illustrated by Figure 9. Trends in advanced multi-mission aircraft are toward high thrust-to-weight ratios - - - approaching 1.0. Subsonic cruise in these aircraft occurs at low levels of drag due to improved aerodynamic efficiency. These two characteristics require the engines to operate at low thrust settings and low airflows. This results in high spillage drag due to the mismatch of engine/inlet flow.

EFFECT OF INLET DRAG ON SPECIFIC FUEL CONSUMPTION

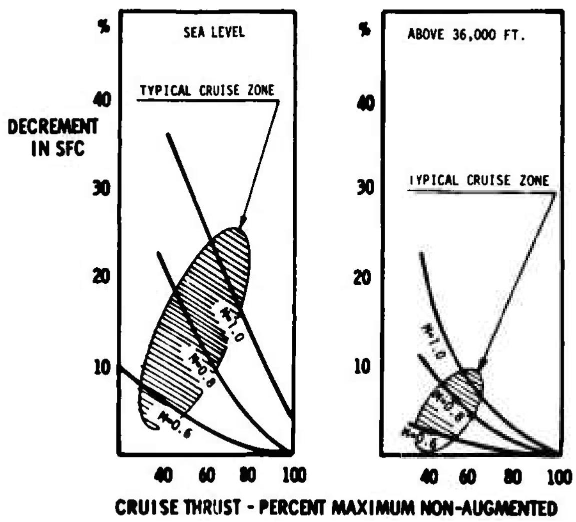

Figure 8

EFFECT OF INLET DRAG ON THRUST

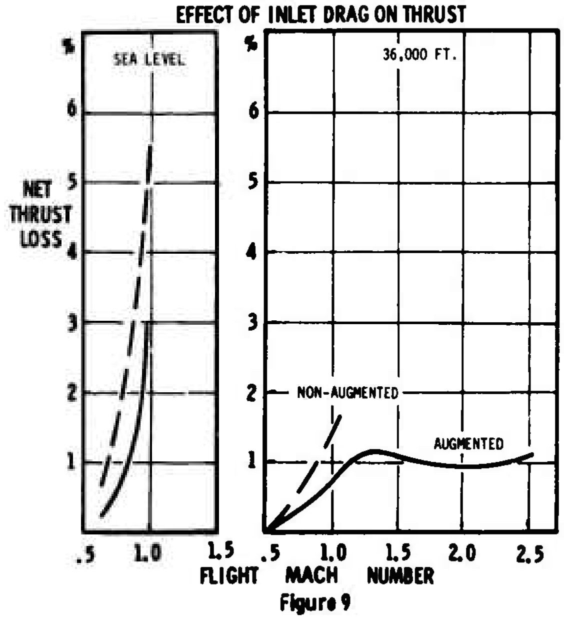

Figure 9

The exhaust system for multi-mission aircraft equipped with twin turbofan engines must consider, in the configuration design, such effects as jet plume interaction and impingement on aircraft surfaces. Further complications arise by virtue of the fact that, relative to a turbojet, the turbofan exhaust nozzle projected area is larger, area requirement (max-min) is greater, pressure ratio is lower.

The external flow over the nozzle can cause high afterbody drags and decrease the nozzle thrust coefficient by virtue of effects on internal gas flow. The nozzle selection therefore requires careful consideration of these factors as well as the upstream boattailing effects of the actual aircraft design. The need for combined aircraft and engine scale model testing to evaluate the effect of the flow field in the vicinity of the nozzle is important and should be done at the earliest practical time.

Aerodynamic compatibility signifies stall free operation of the engine over a wide range of flight conditions. Past practices have been to design the engine components (e.g., fan, compressor, control system), with stability margin based upon past experience and assume that the aircraft inlet would supply acceptable distortion characteristics. Recent problems with multi-mission

aircraft have demonstrated that is approach is not adequate. The stability margin requirements for the engine must be established jointly by the engine and aircraft manufacturers before detailed design starts. These requirements are then assessed and refined as data becomes available from component and full scale testing.

The specific environmental conditions that the engine will be subjected to must also be considered. An advanced fighter aircraft engine may encounter gun gas, rocket exhaust or steam ingestion. By recognizing these problem areas during the initial design phase, aircraft configuration and engine design features can be defined to eliminate ingestion or minimize the effects.

The key steps to achievement of compatibility can be summarized as follows:

- Recognition of potential problem areas (both engine and aircraft) and the voids that exist in available data.
- Clear delineation of responsibilities between the contractors and with the Government.
- Initiation of the program with a design for the airframe and the engine which have a reasonable level of inherent stability margin.
- Identification of key program decision dates with joint integrated test programs to provide data to permit the decisions.
- Extensive, early analysis and testing, plus flexibility, on both sides of the interfaces between the engine and the aircraft to readily adjust for unforeseen problems.
- Concise specification of the role and missions of the planned weapon system.
- Open and frank working arrangements between the engine and system contractors management which directs the activities to achieve compatibility in the most economical and expeditious way.

Final Selection of Technology

The technology selected for the engine qualification program must be consistent with the levels developed in the technology improvement programs. Care must be exercised in the selection. Typically, advances in technology are first demonstrated in a laboratory atmosphere with a configuration designed such that there is little or no compromise to the achievement of the goal. The next logical step is to adapt that technology for inclusion in a development engine. Redesign is usually necessary since the environment has changed from one that is highly controlled (the laboratory) to one (an engine) where the conditions have operational variations that may exceed those experienced in the laboratory. The resulting design changes may well reduce the performance levels achieved.

Also required is an assessment of the technology relative to manufacturing considerations. Since manufacturing cost was secondary during the research and development phases, design changes are to be expected to allow the end product to be made economically.

The reliability and durability aspects, as well as the technology versus schedule risks, must be weighed.

All the facets of the program must be considered in the final selection of technology level and compromises made where necessary.

Technology, as used here, connotes aerothermo as well as mechanical design, material and manufacturing process.

- Aerothermo technology involves the levels of component performance, the choice of the cycle parameters plus the consideration of compatibility requirements.
- Mechanical design technology refers to the configuration of the engine such as number of rotor bearings, number of frames, rotor construction.
- Material technology involves the type of material used in the design such as advanced titanium or nickel base super alloys, filament and plastic composites.
- Manufacturing process technology includes, among others, unique welding, diffusion bonding, or other joining methods.

Obviously, each of these technologies is dependent upon the others to some degree. Risks associated with a challenging cycle selection must be weighed against the material required to achieve a sound mechanical design, and in turn upon the manufacturing process to use that material.

To illustrate this interdependence of technologies, let us consider a basic parameter, engine weight. Since multi-mission aircraft are relatively short range and limited in size, great emphasis is placed on propulsion system weight. In an augmented turbofan engine, the physical size of the engine is determined by the fan airflow and fan pressure ratio required to achieve the thrust specified for each mission point. Having established the low pressure system size, the significant variable remaining relative to engine weight is core size. Core size is directly related to cycle temperature. Figure 10 exhibits the effect of cycle temperature on engine weight. It is clear that the highest possible cycle temperature is desired. The decision, however, must consider the cooling concept, materials and manufacturing processes available.

Program success is dependent, to a great degree, on a well balanced choice of demonstrated technology level, that will produce a reliable, advanced system, in a reasonable time period, at a justifiable cost.

Engine Size Selection

The technology level having been chosen, including cycle parameters, mechanical design, materials and processes, the final decision to be made is the specific size of the engine.

Engine size is established by assessing critical installed thrust requirements at key operating points throughout the flight envelope. Figure 11 illustrates some of these requirements plotted as a function of engine thrust or size and bypass ratio. These curves are for a given level of technologies. The intersection of the most severe augmented thrust requirement (dictated by either a combat point or maximum ceiling) and the most severe non-augmented thrust requirement (usually dictated by cruise ceiling) defines the minimum size engine to meet all requirements.

Engine size, for range requirements, is determined in a similar manner considering the various missions to be performed as shown on Figure 12.

EFFECT OF CYCLE TEMPERATURE ON ENGINE WEIGHT

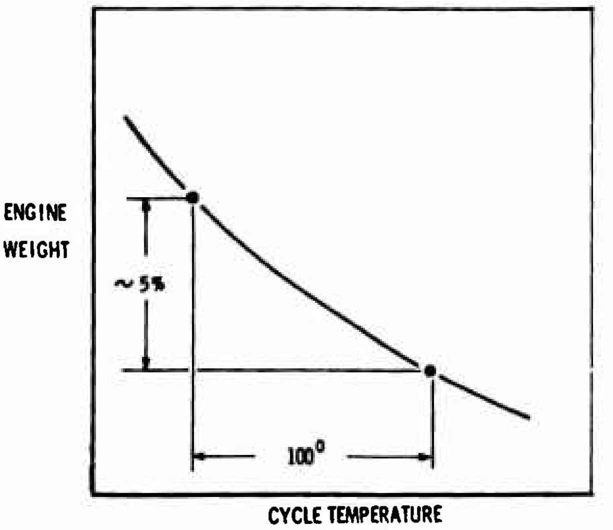

Figure 10

CRITICAL SIZING REQUIREMENTS - THRUST

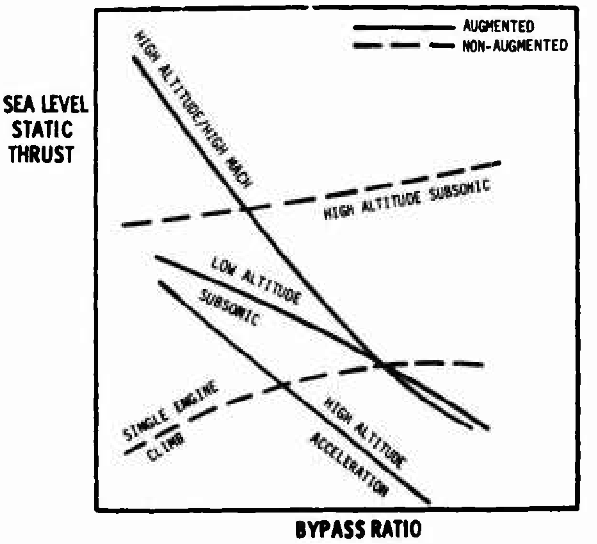

Figure 11

CRITICAL SIZING REQUIREMENTS - RANGE

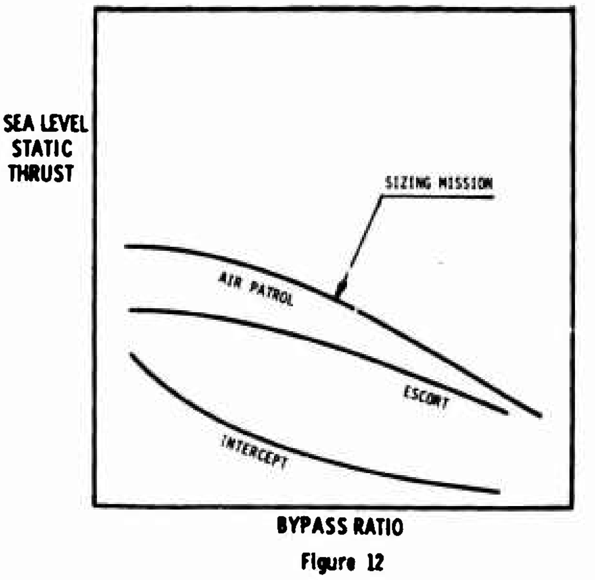

Figure 12

Combining the thrust sizing and range requirement reveals that, in general, multi-mission aircraft which favor high levels of performance have engines oversized for optimum range. Figure 13 shows this and points out that, as in many things, conflicting requirements demand compromise and that some requirements must be more severely compromised than others.

CRITICAL SIZING REQUIREMENTS - RANGE & THRUST

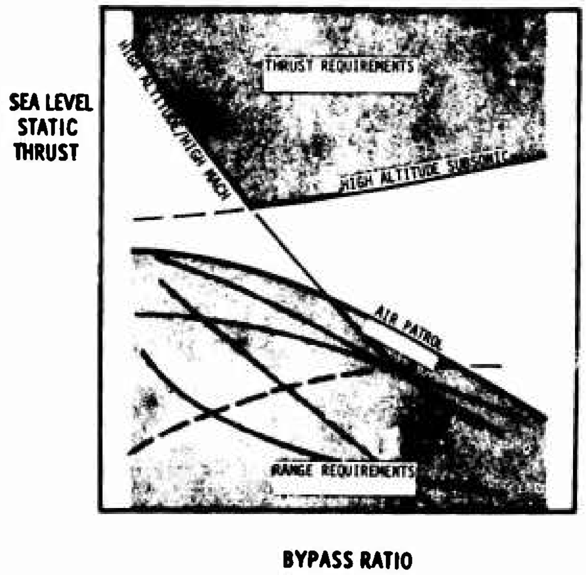

Figure 13

Summary

The development of a modern multi-mission fighter engine requires a weapon system design approach, with close integration of the engine and aircraft manufacturers, to assure a successful program. I have briefly discussed some of the key decision points that are vital to such a program. The criteria for full program initiation is a total system defined by -

- Firm specifications.
- Definitive interface agreements.
- Verified technical approach.
- Firm schedule and cost.

These criteria can be satisified by the proper system design process.

L'INTEGRATION DU PROPULSEUR ET DE LA CELLULE SUR UN AVION DE COMBAT SUPERSONIQUE

par

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L'INTEGRATION DU PROPULSEUR ET DE LA CELLULE

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Sommaire

L'intégration du propulseur et de la cellule amène l'avionneur à étudier des compromis entre des exigences très diverses.

Les idées directrices et le processus de développement sont exposés, et illustrés par l'analyse de la conception et de l'évolution des entrées d'air et arrière-corps des avions DASSAULT. Les contraintes de dessin sont également illustrées par des exemples.

Les méthodes d'essais en vol et en soufflerie sont présentées, ainsi que les premières tentatives de corrélation à l'aide de modèles mathématiques.

I - AVANT-PROPOS -

- Un avion de combat moderne est un compromis entre des exigences très diverses de performances, de manoeuvrabilité et d'utilisation opérationnelle. Ce compromis doit être réalisé dans le souci d'obtenir la meilleure efficacité dans les missions types imposées au départ du projet. Il apparaît particulièrement difficile dans le cas de l'intégration moteur - cellule sur un avion supersonique dont le domaine de vol dépasse Mach 2, et qui doit être efficace non seulement comme intercepteur, mais aussi en mission d'attaque au sol.

- Cette intégration est un problème d'adaptation qui peut être subdivisé schématiquement en trois parties :

  . Adaptation de l'entrée d'air au propulseur.

  . Adaptation de l'arrière-corps - tuyère et fuselage arrière - aux conditions d'échappement du jet propulsif.

  . Adaptation de la ventilation aux besoins du propulseur et de la tuyère.

  Ceci sur toute la gamme des conditions de vol.

- Cet exposé a pour but de présenter :

  . les idées directrices,

  . le processus de développement,

  . les contraintes diverses,

  et de les illustrer par l'exposé de la conception et de l'évolution des entrées d'air et arrière-corps aux Avions Marcel Dassault.

2 - LES IDEES DIRECTRICES sont les suivantes :

- La première consiste à chercher une solution simple et l'améliorer sur les avions suivants en poursuivant parallèlement l'expérimentation en vol et en soufflerie.

C'est cette continuité de la technique qui permet d'aboutir avec le maximum de sécurité à un compromis satisfaisant sur l'ensemble du domaine de vol. Nous évitons donc tout changement radical d'orientation qui conduirait à une mise au point longue et aléatoire. Ce principe sera illustré par le développement des entrées d'air et arrière-corps.

- La seconde consiste à éviter la tentation de se placer trop près d'un optimum qui ne peut être bien défini qu'en envisageant un seul des aspects du problème. Nous en verrons une application dans le problème des pièges à couche limite. Ce principe exclut en particulier la conception d'avions spécialisés, optimisés pour des conditions de vol déterminées, par exemple d'un avion conçu pour Mach 3 qui ne pourrait y arriver du fait de ses caractéristiques transsoniques.

- Ces principes vont être maintenant illustrés par la conception et le développement des entrées d'air et arrière-corps des avions de combat DASSAULT.

3 - LES ENTREES D'AIR -

- La principale fonction des entrées d'air est de fournir au propulseur le débit d'air qu'il exige en assurant :

a) Un bon rendement, donc de faibles pertes de pression génératrice.

b) Une distorsion acceptable par le moteur, c'est-à-dire une pression statique et une vitesse aussi uniformes que possible à l'entrée du compresseur.

c) Un niveau de turbulence suffisamment faible à l'entrée du compresseur.

Ces trois paramètres définissent la qualité du débit d'air fourni au propulseur, les deux derniers s'exprimant sous forme de critères variables avec le motoriste, suivant le mode d'intégration choisi.

Il faut de plus :

d) Assurer une marge suffisante par rapport au phénomène de buzz sur l'ensemble du domaine de vol, dans l'intervalle de températures prévisibles.

e) Obtenir une traînée additive suffisamment faible.

f) Un bruit aérodynamique de niveau acceptable.

- La solution simple qui a été adoptée par les Avions Marcel Dassault tout au début de la conception des MIRAGE III consiste à placer deux prises d'air semi-circulaires à noyau monoconique sur les flancs du fuselage, à une distance suffisante pour assurer l'évacuation de la couche limite dans les conditions de vol les plus défavorables (vues n° 1 et n° 2).

- Cette disposition a été conservée non seulement sur toutes les versions du MIRAGE III et du MIRAGE IV, mais aussi sur les prototypes :

  . BALZAC V et MIRAGE III V à décollage et atterrissage verticaux.
  . MIRAGE III T : 1er avion d'essais de turboréacteur à double flux et à post-combustion.
  . MIRAGE F 2 : Avion d'attaque au sol.
  . MIRAGE F 1 : Avion d'interception et d'attaque au sol.
  . MIRAGE G : Avion polyvalent.

- L'évolution de ces entrées résulte d'une longue série d'essais en vol et en soufflerie :

  . Plus de cent configurations de pièges - d'entrées d'air auxiliaires - de dispositifs internes - ont été essayées en soufflerie.

. En vol, les essais ont porté sur :

Les pièges à couche limite : 12 configurations.

Les entrées d'air auxiliaires : 7 configurations.

Les dispositifs internes : cloison de séparation, prélèvement autour du fond de marche pour la ventilation.

Les lèvres des entrées principales : un avion a volé avec des lèvres épaisses.

- Le développement a porté surtout sur les points suivants :

. L'aérodynamique des pièges à couche limite a été améliorée. Ces pièges comportaient initialement un toit qui reliait la partie supérieure de l'entrée au fuselage ; l'expérience a montré que ce toit nuisait au bon fonctionnement du piège en réduisant le débit d'évacuation de la couche limite et en créant des chocs supplémentaires interférant avec les chocs dûs à l'entrée. Ce toit a donc été supprimé (vue n° 2).

La largeur du piège est évaluée, pour chaque avion, à partir de calculs de couche limite et de règles pratiques découlant de l'expérience acquise en vol et en soufflerie. Si l'on se plaçait du seul point de vue du bilan poussée - traînée, il serait possible de réduire légèrement cette largeur, mais il est probable que cette réduction entraînerait des troubles tels que instabilité de fonctionnement et niveau de turbulence trop élevé.

. La forme des noyaux monoconiques a été améliorée, le but étant surtout d'éviter un décollement au pied du choc droit qui se forme au voisinage de l'entrée.

D'autre part, des noyaux biconiques ont été essayés en soufflerie et en vol ; cette disposition, qui permet d'atteindre des Mach plus élevés, est en cours de développement.

. La loi de déplacement du noyau était à l'origine très simple, le déplacement étant une fonction linéaire du Mach, sans faire intervenir la température ; nous avons conservé le principe d'un programme fonction du Mach seul, mais la loi actuelle se traduit par une succession de segments qui permettent de mieux s'approcher de l'adaptation de l'entrée aux conditions de vol, tout en conservant une marge suffisante par rapport à la limite de buzz dans la plage de températures possibles.

. Les entrées d'air auxiliaires ont été aussi fortement améliorées en travaillant leur position en profondeur et en hauteur, ainsi que le détail de leur dessin (photo n° 3). Du fait des contraintes résultant de la structure particulière à chaque avion, leurs dimensions diffèrent d'un avion à l'autre, mais elles se ressemblent beaucoup sur les avions d'une même génération. Ces entrées auxiliaires permettent actuellement d'atteindre au cours du décollage un rendement très honorable pour des entrées à lèvres minces, ceci avec une faible distorsion à l'entrée du compresseur.

. D'autre part des essais en vol ont montré que des lèvres épaisses apportent un gain de rendement au cours du décollage, mais une perte en supersonique. Nous avons donc conservé des lèvres minces.

. Enfin l'expérience a montré qu'une cloison séparant les débits droite et gauche jusqu'à l'entrée du compresseur n'est nécessaire que sur les entrées courtes du type MIRAGE III.

Nous n'indiquons pas les valeurs de rendement obtenues dans les différentes phases du vol, parce que ces chiffres ne représentent qu'un aspect de la réalité, car l'efficacité d'une entrée d'air dépend aussi de sa traînée additive - de même que l'efficacité d'une tuyère ne dépend pas que de son rendement, mais aussi de sa répercussion sur la traînée externe - l'essentiel est que, globalement, nos avions soutiennent la comparaison avec les avions concurrents alliés ou russes.

En conclusion, le développement de ces entrées d'air montre qu'une solution simple, mais perfectionnée de façon continue, a permis d'obtenir un compromis satisfaisant entre les diverses exigences d'alimentation en air d'un avion de combat à Mach 2.

4 - LES ARRIERE-CORPS -

- Nous abordons maintenant le problème de la conception et du développement de l'arrière-corps, c'est-à-dire de la tuyère et du fuselage arrière.

  Ces deux éléments, qui sont matériellement bien distincts, sont reliés sur le plan des performances par les interactions entre le propulseur et l'avion, et éventuellement par la ventilation, si elle est réalisée par des entrées auxiliaires.

- La tuyère, qui est sous la responsabilité du motoriste, mais qui fait l'objet d'une collaboration avec l'avionneur, doit satisfaire aux exigences suivantes :

  . Adapter la section du col aux conditions de fonctionnement du moteur, avec ou sans post-combustion.

  . Minimiser la traînée de culot. Ce terme mérite une définition ; aux Avions Marcel Dassault, il représente la variation de l'écart poussée moins traînée par rapport à ce que serait cet écart s'il n'y avait pas d'interactions entre le propulseur et l'avion.

  Il comprend donc :

a) L'écart de poussée, par rapport à la valeur donnée par le motoriste, qui correspond à l'intégrale des pressions relatives établies à l'intérieur du culot.

b) La traînée due à la ventilation.

c) La variation de traînée externe due aux interactions du jet propulsif avec l'écoulement externe.

- De plus la tuyère et son mécanisme doivent fonctionner correctement dans tout le domaine de vol, ce qui suppose que les températures et les efforts ne dépassent pas les limites admises.

- Enfin, l'écoulement doit être suffisamment stable pour ne pas engendrer de buffeting ou de mouvement brutal de l'avion, ce qui exclut tout amorçage ou désamorçage brutal.

- L'ensemble de ces exigences - et en particulier la nécessité d'obtenir un compromis acceptable entre le vol subsonique sans post-combustion et le vol supersonique avec post-combustion - a conduit la SNECMA à concevoir une tuyère à volets chauds et volets froids articulés et reliés mécaniquement par des bielles (vue n° 4). La tuyère se compose donc en fait d'une tuyère chaude qui règle la section de col, et d'une tuyère dite froide qui agit sur le taux de détente du jet, les deux étant du type multipétales. L'ensemble est actionné par des vérins répartis autour du canal de post-combustion (vues n° 5 et 6).

- La section de col au régime plein gaz sec étant beaucoup plus faible qu'avec post-combustion, il s'ensuit que le culot géométrique serait très important en l'absence de volets froids, en vol subsonique - ce culot étant l'écart entre la section du fuselage au bord de fuite et la section du col. L'addition de la tuyère froide réduit le culot en prolongeant le rétreint du fuselage, et permet d'obtenir une traînée de culot acceptable, à condition d'éviter le décollement de l'écoulement externe sur les volets froids.

- Le dessin du fuselage arrière doit permettre de raccorder le maître-couple du fuselage à l'emplanture des volets froids de façon aussi continue que possible, et sans rétreint exagéré. De plus, il faut loger les équipements et la structure de la zône arrière, en particulier les cadres d'attache des empennages. Le raccordement avec la tuyère froide se fait par l'intermédiaire d'une membrane souple appelée "marguerite" qui assure le jeu nécessaire à la dilatation en maintenant une bonne continuité.

- Enfin, la ventilation doit être suffisante pour éviter un échauffement excessif de la zône arrière, en particulier des vérins de commande de la tuyère et du canal de post-combustion. De plus elle doit être adaptée aux conditions d'écoulement dans le culot de façon à se rapprocher de la traînée de culot minimum.

- La mise au point de cet ensemble a nécessité un grand nombre d'essais en vol et en soufflerie.

Ces essais ont eu pour but :

a) D'une part, de mieux comprendre les phénomènes d'interaction entre les écoulements interne et externe, et le rôle de la ventilation.

b) D'autre part, de définir des règles simples de dessin et d'adaptation de la tuyère froide et du fuselage arrière, et d'orienter le développement ultérieur.

Ils ont permis en particulier :

- De se rendre compte qu'un décollement externe en supersonique n'entraîne pas nécessairement une augmentation de la traînée de culot.

- De définir une règle simple permettant d'éviter le décollement externe en subsonique.

- En vol, les essais ont porté sur 24 configurations différentes de volets froids régulés ou de viroles fixes - c'est-à-dire de tuyères froides rigides.

- En soufflerie, près de 30 configurations ont été essayées en subsonique - 15 en supersonique, plus 20 actuellement en cours - (vue n° 7).

- La méthode et les moyens mis en oeuvre pour exploiter et relier ces essais sont exposés dans la partie consacrée au processus de développement - mais il faut signaler dès maintenant que la corrélation des résultats obtenus en vol et en soufflerie n'est pas toujours bonne.

Il paraît probable que les principales raisons de ces écarts entre soufflerie et vol soient les suivantes :

a) La couche limite en bout de fuselage est très différente sur la maquette de celle qui existe sur l'avion.

b) Les températures du jet, de la ventilation, et de l'écoulement externe ne sont pas reproduites en soufflerie.

c) Les maquettes utilisées ne représentent que la partie arrière de l'avion ; de plus elles sont de révolution - alors que le fuselage de l'avion ne l'est pas.

d) Il est difficile d'éviter toute interaction du montage de soufflerie sur l'écoulement de culot, et en particulier sur le sillage.

5 - LE PROCESSUS DE DEVELOPPEMENT -

- Nous abordons maintenant le processus de développement, c'est-à-dire la méthode et les moyens mis en oeuvre pour aboutir le plus rapidement possible à une mise au point satisfaisante des entrées d'air et arrière-corps.

5.1 - Parlons d'abord des essais en vol -

La méthode utilisée vise à obtenir des résultats significatifs dans le délai le plus court. Pour cela nous cherchons à gagner du temps sur les fabrications, la réalisation des essais, et leur exploitation.

- Ainsi pour réduire les délais et coûts de fabrication, de nombreux essais de tuyères ont été réalisés avec des viroles fixes qui représentaient la configuration de la tuyère réelle régulée dans certaines conditions de vol.

- D'autre part, nous ne réalisons pas 12 prototypes, par exemple, pour étudier un avion, mais nous menons une expérimentation continue de tous les dispositifs à mettre au point sur les prototypes successifs d'avions divers tels que MIRAGE III A, MIRAGE III T, MIRAGE F 2 et F 1, MIRAGE G.

  Ces prototypes sont constamment modifiés, et l'on profitera, par exemple, des essais d'un système d'armes pour essayer également une tuyère fixe, ou une configuration de piège à couche limite. Ceci permet de réaliser les essais dans le meilleur délai.

- Enfin la méthode d'exploitation par ordinateurs en temps réel permet d'obtenir très rapidement les résultats sous la forme la plus élaborée.

L'organisation est la suivante :

a) A bord de l'avion, les instruments de mesure - gyromètres, accéléromètres, capteurs de pression et de vibrations, strain-gages, etc... - transmettent leurs informations d'une part à des enregistreurs magnétiques - d'autre part directement à un ordinateur au sol par un système de télémesure.

b) Au sol :

  - Pendant le vol, certains des paramètres sont visualisés et traités par l'ordinateur, enregistrés sur bande magnétique - une partie étant imprimée à cadence rapide.

  - Après le vol, les bandes magnétiques enregistrées sur l'avion sont traitées par l'ordinateur pour restituer sous la forme la plus utilisable l'ensemble des informations recueillies.

- Les mesures faites pour la mise au point des entrées d'air et arrière-corps portent sur des pressions, des consommations kilométriques en palier, des températures d'impact, des paramètres géométriques tels que positions des volets chauds et des noyaux d'entrée d'air, et des explorations de couche limite.

5.2 - Parlons maintenant des essais en soufflerie -

Ils visent deux buts :

- D'abord, reproduire en soufflerie les configurations essayées en vol - mais avec les différences signalées précédemment - pour établir des comparaisons qui - ainsi que je l'ai déjà dit - ne sont pas toujours bonnes.

- Ensuite, étudier de nouvelles configurations que l'on suppose capables d'apporter des améliorations.

  La maquette est alimentée en air comprimé par l'amont et des vannes permettent de régler la pression génératrice de l'écoulement primaire - c'est-à-dire du jet - et de l'écoulement secondaire - c'est-à-dire de la ventilation.

Nous mesurons :

. les débits primaire et secondaire
. les pressions internes et externes sur l'arrière-corps
. les efforts sur différents éléments tels que fuselage arrière, tuyère froide
. la couche limite en bout du fuselage.

Nous pouvons donc faire certains recoupements entre efforts et intégrales de pression - évaluer une traînée de culot - puis comparer aux pressions et à la traînée de culot déduite des mesures en vol.

5.3 - Nous avons déjà insisté sur le fait que les recoupements vol-soufflerie ne sont pas toujours bons.

La décision que nous avons prise pour remédier à ce défaut de corrélation consiste à investir dans le domaine théorique pour arriver à un modèle mathématique qui permette de retrouver avec une approximation suffisante les résultats d'essais en vol et en soufflerie. Cette méthode nous paraît la seule valable pour interpréter ces essais, c'est-à-dire pour en tirer des conclusions sérieuses.

Ces modèles sont actuellement en cours de développement. Ils comprennent :

a) Pour les entrées d'air, un programme de calcul de l'écoulement non visqueux par la méthode des caractéristiques, et un programme de calcul des couches limites.

b) Pour les arrière-corps, un programme de calcul d'un culot non amorcé en vol supersonique et un programme de calcul d'éjecteurs.

Ces derniers programmes tiennent compte des interactions visqueuses entre les écoulements internes et externes. Ils se basent sur les théories du mélange turbulent isobare et du recollement turbulent supersonique.

Ils permettent également de calculer un éventuel décollement de l'écoulement externe dû au choc de bord de fuite, et d'analyser ses conséquences sur la traînée de culot.

C'est ainsi que nous avons constaté qu'un décollement externe en supersonique n'est pas forcément nuisible.

Ces programmes ont été testés sur certaines des configurations essayées en vol et en soufflerie, et les résultats sont assez encourageants.

Néanmoins, les théories existantes ne permettent pas d'étudier un certain nombre de problèmes tels que :

a) L'écoulement autour d'un culot en vol subsonique ou transsonique.

b) Les décollements en vol subsonique sur un fuselage ou une entrée d'air, en particulier leur structure.

c) La détermination de la frontière d'un jet non visqueux issu d'un convergent tronconique.

d) La zône de recollement supersonique sur une paroi solide ou fluide - et ce qui se passe quand le recollement est perturbé - ce qui paraît le cas quand on a affaire à un éjecteur court.

Nous souhaitons donc que les spécialistes de ces questions arrivent à construire des théories qui permettent d'aboutir à un modèle représentant ces phénomènes avec une précision suffisante.

Nous insistons en particulier sur les problèmes que posent les culots en vol subsonique et transsonique, où l'absence d'outils de travail théorique complique la tâche de l'avionneur, en affectant les prévisions d'une forte marge d'erreur.

6 - LES CONTRAINTES -

- Nous abordons maintenant l'exposé de certains facteurs limitatifs ou impératifs qui amènent l'avionneur à des compromis, ou à des choix, dans l'intégration du propulseur et de la cellule et qu'on peut appeler des contraintes.

  Les principales contraintes restreignant la liberté de dessin des entrées d'air et arrière-corps sont imposées par l'organisation et la structure de l'avion.

  Ainsi le logement du train d'atterrissage principal dans le fuselage complique le dessin des canaux d'entrée d'air, qui guident la diffusion depuis l'entrée proprement dite jusqu'au compresseur.

- De même les cadres d'attache des empennages horizontaux et verticaux imposent des contraintes au dessin de la partie arrière du fuselage.

  Nous nous sommes efforcés de réduire le diamètre extérieur de ces cadres, afin d'éviter de dessiner un fuselage arrière trop rétreint, qui pourrait dans certaines conditions de vol, amener une augmentation de la traînée de culot.

- Mais les impératifs de la sécurité peuvent aussi dicter certains choix - par exemple, la ventilation de la zône arrière du réacteur était assurée autrefois par des prises d'air auxiliaires, appelées écopes - or l'expérience a montré qu'en cas de fuite de carburant, il pouvait arriver que du pétrole soit avalé par ces écopes, ce qui entraîne des risques d'incendie. Nous avons donc remplacé les écopes par des prises de ventilation sur la manche d'alimentation du réacteur, autour de l'entrée du compresseur - alors que sur certains avions, les écopes sont mieux adaptées si la pression du culot reste peu élevée dans tout le domaine de vol.

- Enfin, les exigences concernant les poids et leurs répercussions sur le centrage nous ont conduit à éliminer certains dispositifs lourds d'arrière-corps, par exemple les "blow-in-door" du TF 30.

7 - CONCLUSION -

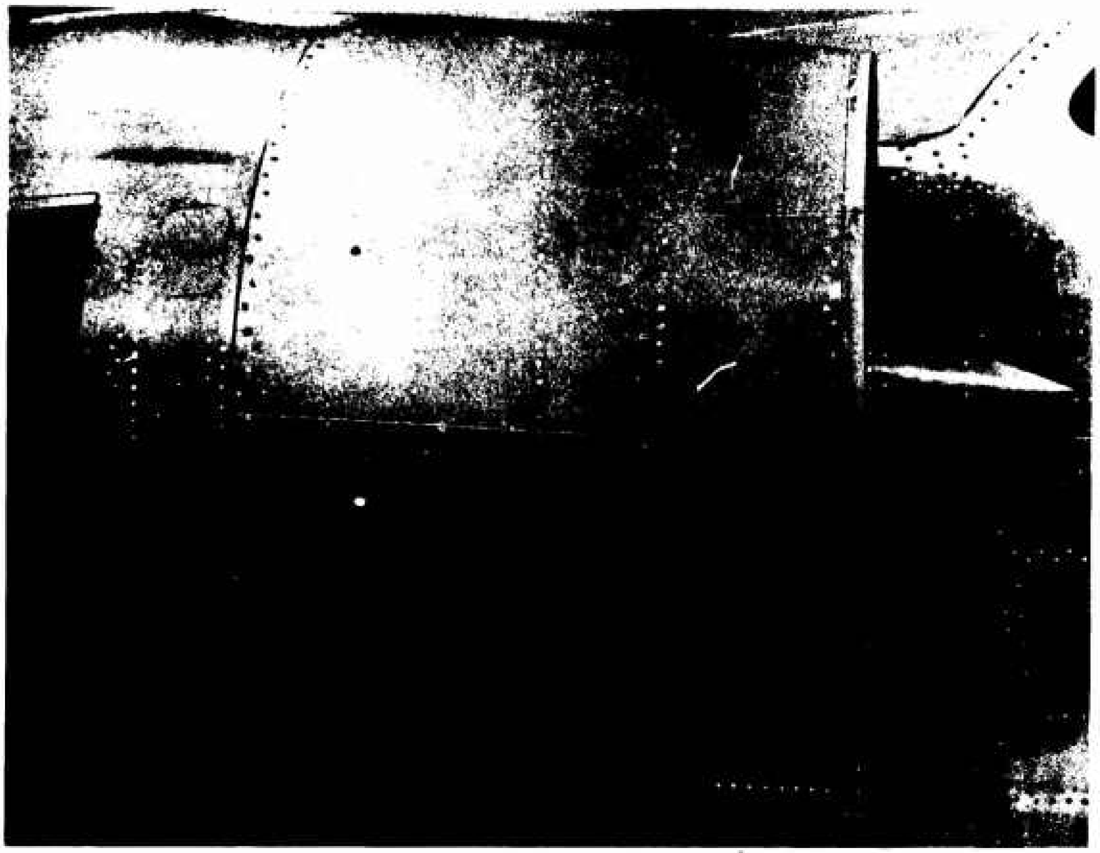
Une méthode d'intégration du propulseur et de la cellule a été présentée, et illustrée par l'analyse de la conception et du développement des entrées d'air et arrière-corps aux Avions Marcel DASSAULT.

Nous ne prétendons pas que cette méthode soit la meilleure, mais son efficacité est démontrée par la rapidité de réalisation et la compétitivité des avions DASSAULT. Nous pensons de plus que les solutions simples qui ont été développées constituent un bon compromis pour un avion de la classe Mach 2 +, sous l'angle d'une efficacité globale dans les missions d'interception, de police du ciel, et d'attaque au sol.

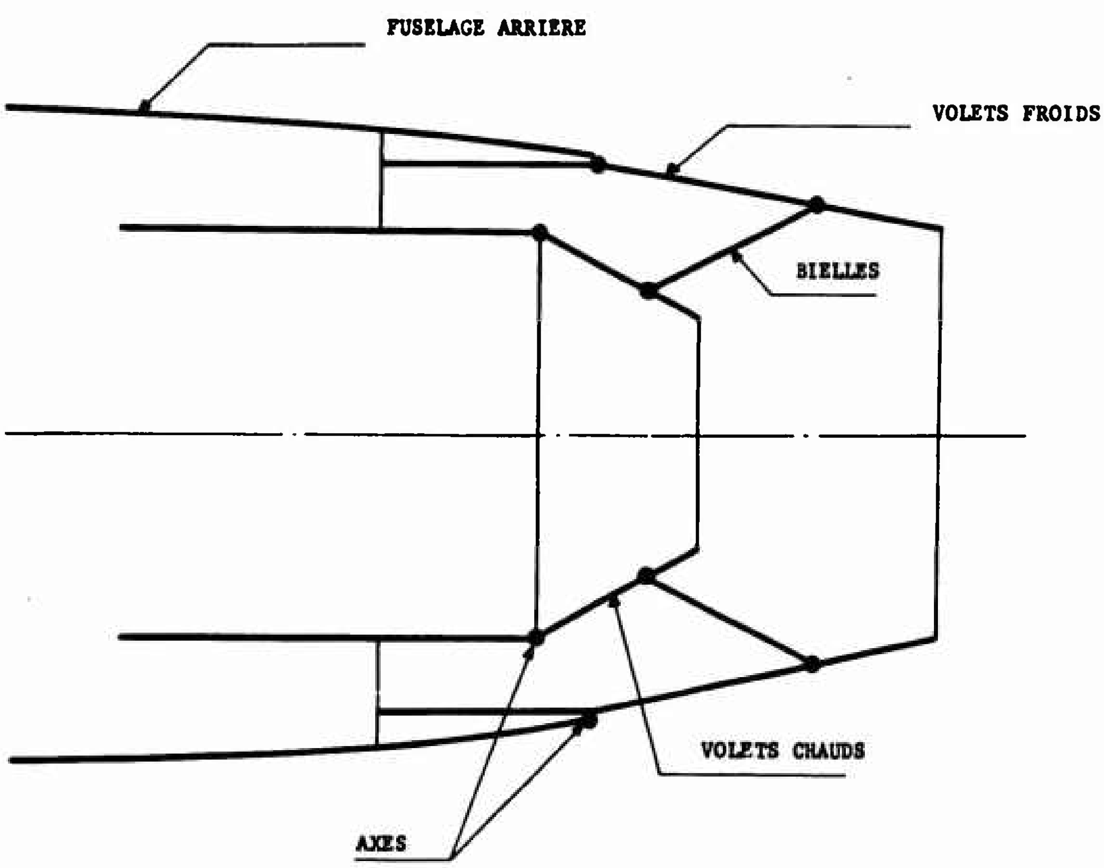
Nous insistons enfin sur le souhait concernant le développement de théories permettant d'étudier l'écoulement sur les arrière-corps en vol subsonique et transsonique.

N° 1 - Entrées d'air vues de face.

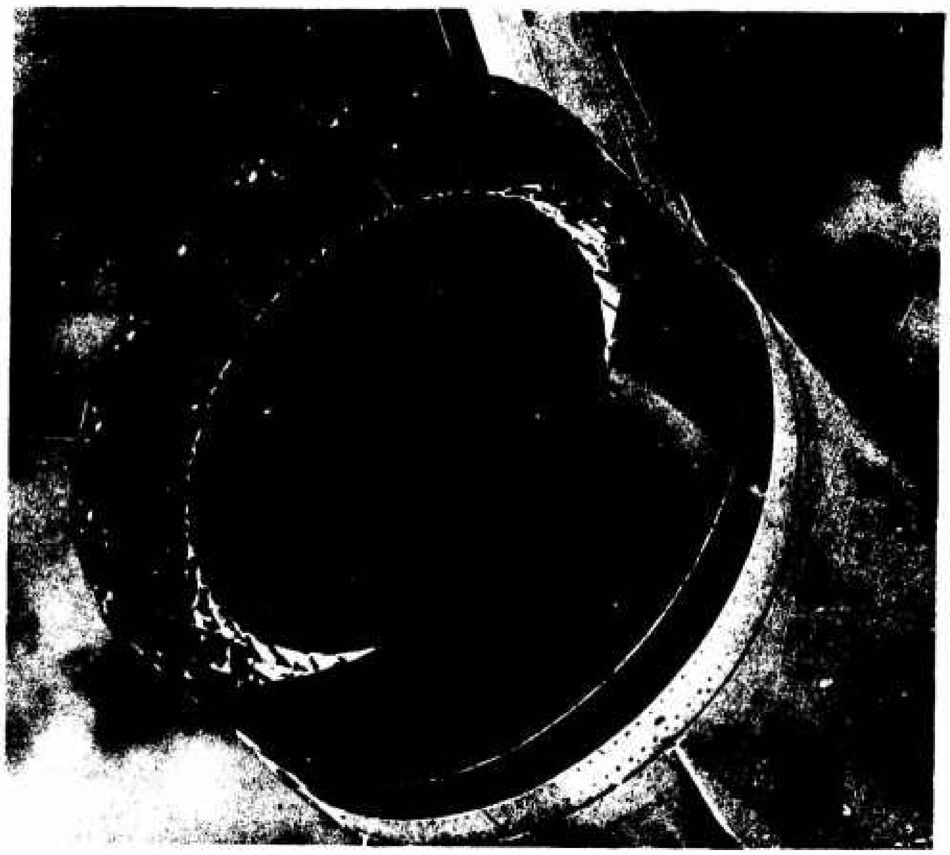
N° 2 - Entrées d'air vues de 3/4 avant.



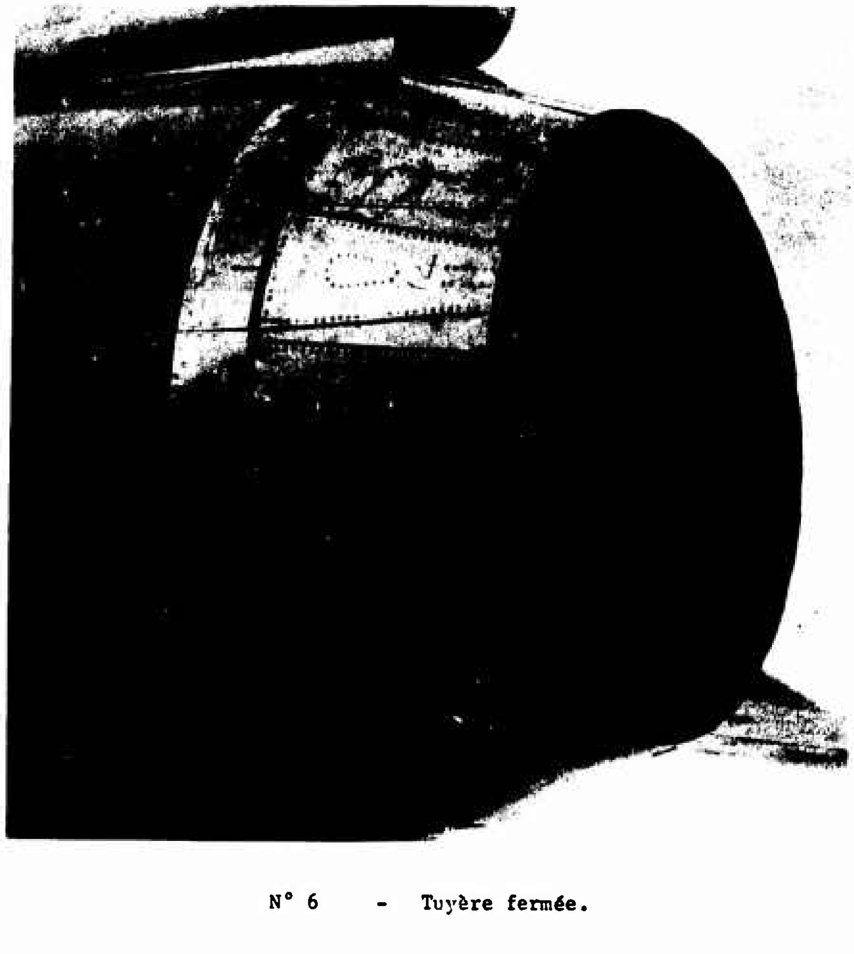
N° 3 - Vue latérale des entrées d'air.



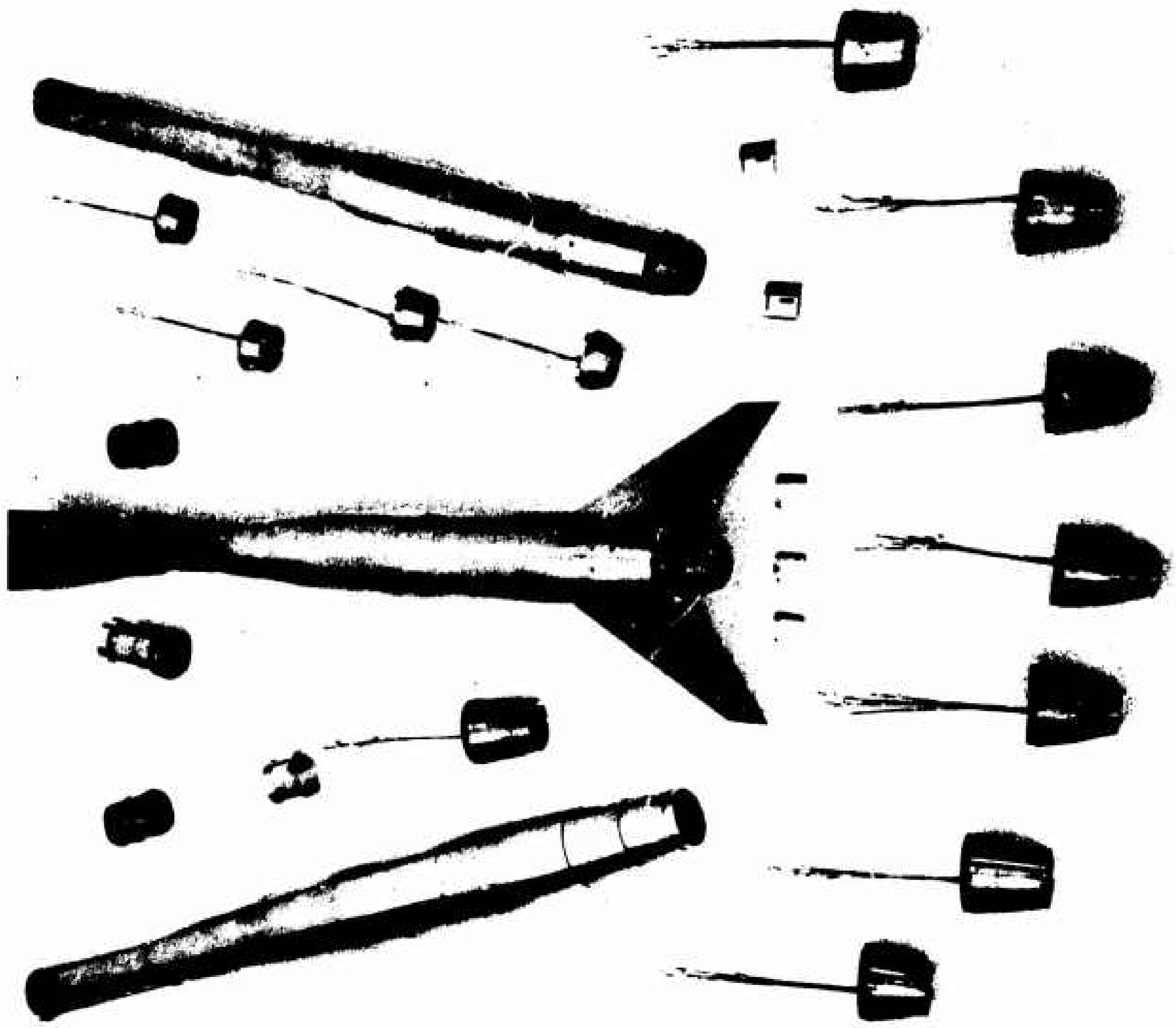
N° 4 - Schéma de la tuyère.



N° 5 - Tuyère en position pleine P.C.



N° 6 - Tuyère fermée.



N° 7 - Quelques tuyères et fuselages essayés en soufflerie.

Designer's View on Structural Problems

by

Rolf Riccius

assisted

by

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Summary

The aircraft designer is obliged by analysing carefully the specific requirements to present an optimum approach to a weapon system within a reasonable number of iterations. Therefore more accurate data are requested from all concerned scientific departments. Especially for optimizing the structure better knowledge of weight prediction, materials, design research and production methods should be available. These problems will be discussed by means of a number of examples, some development trends will be demonstrated and in summing up recommendations will be issued pointing out in which way and by what means preliminary design with special attention to structural design should be improved.

Sommaire

L'ingénieur réalisant des projets a l'obligation d'analyser avec soin les caractéristiques exigées et ensuite, par des quelques procédés itératifs, de soumettre un système d'armes qui présente l'optimum. Des informations plus exactes de tous les domaines d'activité – en particulier sur le domaine du calcul de masse, des études du matériel, des recherches de construction et de la production– sont impératives pour l'optimalisation de la structure. Ces exigences sont discutées au moyen d'une suite des exemples, et quelques tendances d'étude sont indiquées. Finalement, des recommandations sont exprimées au sujet de la maniére et des moyens perfectionnés qui devraient servir à approfondir les travaux de projet.

Acknowledgement

The author is indebted to the management of Vereinigte Flugtechnische Werke GmbH, Bremen, for approval to deliver this paper.
He further wishes to express his sincerest thanks to the members of his staff and to the gentlemen listed on the front page for their valuable help and support.

1. Introduction

Within the scope of this paper and in adherence to the given theme I shall try to define the requirements based around the structural design of an aircraft and to establish some guiding principles foı realization of military aircraft. In doing so, emphasis will be placed on the design of fighter-type aircraft.

Fundamentally the requirement to be met by the aircraft structure is the following one :

> The structure must withstand the specified ultimate loads and it must take the loads imposed in operation during its scheduled life without fatigue (1).
>
> The weight of thr structure must be kept at a minimum which can only be achieved by finding an optimum structural lay-out and by loading all members of the structure as specified. Optimum cross-sections should be . ılected for stiffness and strength reasons and the structural design should avoid stress concentrations to meet the specified fatigue life with minimum weight imposed.

Independent of their specific performance requirements, advanced fighter-type aircraft have some general features in common. They are aircraft with an average life time of 3000 to 6000 hours which in consideration of their maneuverability, are subject to high g-loads (6 g to 8,33 g) especially at low level, are subject to severe gust loads, and consequently are at a low stress level when cruising. During their mission most of these aircraft are flying at transonic and/or supersonic speeds and their skin therefore is subject to kinetic heating. On account of the high thrust installation especially in STOL or VSTOL configurations with integrated thrust reverser or thrust deflection and also when guns are fired, these aircraft are subject to severe sonic loads.

Fighter-type aircraft are expected to operate at a high degree of cost effectiveness. They therefore require being designed for smallest wetted area, maximum utilization of available volume and high installation density, but at the same time excellent accessibility for maintenance purposes should be provided.

Hence it follows that the structure of such an aircraft is temporarily subject to high stresses, heat loads and sound admission, and that in consideration of service life and high installation density this structure should be made being reliable and maintenance free or at least nearly so.

2. Mission analysis and load cri:eria

A careful analysis of the specified mission profile will supply the design engineer with information about the number and magnitude of g-loads the aircraft has to withstand, and where local stress-concentrations have to be assumed, e.g. caused by gun firing or weapon delivery. A typical fighter-type aircraft load spectrum given in fig. 1 for a flight time of 1000 hours at sea level shows e.g. the expected load cycles leading to fatigue expressed as load factors plotted against cumulative frequency.

Naturally these combined gust and maneuver loads, to which crew and aircraft are exposed, will increase as the flight level of the mission is lower, the speed is higher, and the wing loading at equal lift characteristic is lower.

At higher speeds depending on the flight level, with VSTOL aircraft also on account of direct jet admission, the skin of the aircraft may be heated up to such a degree that the strength of the material is reduced.
Fig. 2 shows two typical examples, one the temperature occuring locally on the structure of an aircraft at vertical take-off in ground proximity, and the other one the skin temperature of an aircraft flying at different flight levels and at variable Mach numbers. Accordingly temperatures up to 430° C at VTO or temperatures from 160° C at $M = 2,2$ to 335° C at $M = 3$ may develop during the mission whatever speed the aircraft is scheduled to fly.

Furthermore sound pressure distributions are to be expected in flight as shown e.g. in fig. 3 for a STOL aircraft wher afterburner are lit or unlit and when guns are fired, or for a typical VSTOL-aircraft according to fig. 4 at conventional, short and vertical take-off. The sound pressure occuring are of a magnitude from 160 to 175 db and they can result in fatigue after a very short time, especially in joints if the structure is not adequately dimensioned or shaped (2).

3. Task of the design engineer

It is the task of the design engineer in cooperation with the project analysis engineer to investigate the military requirements and to select specified aircraft qualities, to establish design objectives and on the basis of these to submit the design for the aircraft in question with only a few iterations. For this purpose he analyses and optimizes the aircraft according to the total system-principle and not, as often done in the past, by dealing with and optimizing aerodynamic or engineering details as isolated problems. During this process specialists often have to tolerate that their particular demands will be abandoned and replaced by compromises in favour of the overall optimum.

The flow of a typical design process according to (3) may be seen in fig. 5 which illustrates in plausible form the flow of design and iteration loops whereas in fig. 6 the flowchart of a structural weight determination of the first level is demonstrated. Aside from the fact that nowadays the design

engineer is capable of carrying out such an optimization calculation in a short time with the aid of a digital computer, the results on one hand are directly depending on the accuracy of inputs, e.g. aerodynamic data and on the other one on the reliability of a number of partial optimization processes, e.g. weight determination. The still existing uncertainty in predicting structural weights of advanced aircraft by considering specific loads like sound, temperature and the progress of material development is still the greatest problem to the design engineer. Even if concededly excellent achievements have been made in this field during the last years, it is here where future efforts should be concentrated.

The subprograms for the weight calculation must be of sufficient variation width so that in case of changes of performance requirements by means of the lay-out program the aircraft may be scaled sufficiently accurate and growth factors (4) (5) (6) can be determinated, which are very important for evaluation of the design.

Since the design engineer is urged to submit a design which upon changes in performance or in case of added weight shows the slightest possible and not an avalanche of effects in return, he must be furnished with accurate and improved methods which enable him to prove his point.
Giving a typical example, in fig. 7 the difference in growth factor quantities can already be seen according to the fact that the aircraft, which fixed weights are varied, are CTOL or VSTOL types. The VSTOL growth factor is greater since weight increase effects a greater number of aircraft components in a more critical manner.

4. Structural optimization and produceability

The structure optimization of military aircraft also must be effected from the total system angle considering especially the requirements for maintainability, reliability, vulnerability and the specific loads mentioned before.

Actual selection of the type of construction strongly depends on the following problems :

1 - load spectrum (which means forces, moments, sound, heat, duration of mission) and stiffness criteria

2 - Produceability effected by

production methods available and production cost,
breakdown into assemblies for logistic reasons and
materials available

3 - installation of equipment and maintainability especially with respect to

routine maintenance free or nondestructive inspection
accessibility

4 - failure probability (which means safe-life or fail-safe to be considered) and vulnerability

Hence it follows that structure optimization and selection of the type of construction must be accomplished with an eye on real disturbances like cutouts for doors and panels, joints, excentricities etc. To this fig. 8 and 9 show a few examples of aircraft the structure of which had to be suitably broken up only to provide access, and in fig. 10 it can be seen by some characteristic sections what installation density often interferes with structural optimization.

It is interesting to note that again and again cutouts, e.g. access panels for maintenance purposes have to be provided after the structural assemblies have been frozen and thus the selected optimum structural design is rendered ineffective. In order to obtain a reliable structure, the principles of fail-safe and safe-life, respectively, naturally must be considered and this with increased strictness if the aircraft on its intended sorties will permanently be exposed to enemy threat.

The continously required struggle for minimum weight taking into account required reliability and that production methods are available is reason to realise that the percentage of integrated parts made by machining, pressing or precision casting is steadily increasing which fact, e.g. is illustrated in fig. 11 with some of the structural components of the VAK 191 B.

However, more concentrated efforts are necessary to develop almost notch-free joints in order to obtain a better utilization of the material. This includes also improved welding methods like electron beam or LASER welding and a foolproof quality control.

As a tight bottleneck in reliable realization the sonic loaded structure still must be considered. Although these are useful and accurate calculation methods (7) for the determination of sound pressure distribution on the ground and in flight available, there is a lack of test results for flat, curved, double curved, stiffened or sandwich skin panels. These results are unobtainable in reasonable time on account of the enormous scope of tests involved. Therefore appropriate calculation methods should be developed. To this day there are not sufficient data sheets for sound loaded structures or corresponding airworthiness requirements available. Regular checks of sound field distribution and determination of sound pressure loads by strain gages and microphones in airborne aircraft are necessary to varify calculated data.

5. Material development

The aircraft designer wants for airframe optimization long-life materials which must be of light weight, flexible, of high strength, temperature resistant, crack resistant, corrosion resistant, and possibly weldable. The basic material now as before is aluminum. This material today has been developed to high static strength values but without attaining contemporarily a significant improvement of its response to fatigue. High-strength aluminum therefore must be developed to better ductility and in improved response to crack propagation and stress corrosion which leads to a longer life before it can be used with a corresponding weight saving.

This naturally applies preponderantly to almost notch-free integral components with high tensile stress levels where an improvement of material life really pays off.

Titanium alloys with good properties in form of weldable sheet metal or forging material or with still improved temperature resistance will be available to a greater extent and result in a reduction of airframe weight. If 30 % of the airframe weight are titanium, the structural weight is reduced by approximately 10 per cent. (8)
As far as hot and cold working as well as weldability and stress-corrosion of titanium alloys is concerned, they must be improved.

With steel and steel alloys continously advanced properties may be expected. Steel showing a static strength up to 200 kp/mm^2 also in forgings will be on the market before long, but the fatigue strength as well should be improved. Highly heat-resistant nickel alloys with good creep behaviour and good deformation behaviour are available and suitable for aircraft speeds up to the equivalent of $M = 7$.

Beryllium at present is an attractive new material, one third less weight than aluminum (8) withstanding up to 430° C without loss of strength, as yet only lacking ductility. The material is rather expensive and its ductility may be improved by fiber-reinforcement.

That points to the probably most promising trend in material development, to composite materials. Best known are glass-fiber, carbon-fiber, and boron-fiber compound materials. Depending on fiber orientation, the weight of the two latter ones may be 15 per cent less than that of aluminum (9) and their strength two times that of titanium alloys. Structural weight savings between 30 and 40 per cent are predicted for the future (10).

While glass-fiber reinforced plastic components today may be looked upon as seldom stressed but well established nonstressed construction elements, the progress of carbon-fiber and boron-fiber compound materials, apart from a few test applications is of some importance only in engine construction but still slow in airframe construction. As example in fig. 12 the carbon-fiber compound material Hyfil, so far well proved in engine construction, is shown with its properties plotted against per cent fiber content (11). Apparently all the difficulties which arose in producing compound materials have not yet been overcome, and it will be in particular the change in production method which will concern some aircraft manufacturers with respect to their park of NC-machines. In addition the application of compound materials requires more engagement in detail design studies especially where joining elements are concerned.

Cause for alarm, however, gives the fact (7) that the behaviour of the materials under sound pressure loads practically is known only for aluminum whereas there exists a practically complete lack of such knowledge for titanium, steel, beryllium and the before mentioned compound materials.

Finally the designer must be interested in the way in which the materials selected by him can be tested and controlled even in processed condition, too. Nondestructive testing of materials and joints and inspection for cracks and corrosion by X-ray or ultrasonic methods are therefore of particular increasing importance. Standardized regulations for recommended test methods are therefore to be established and should be observed already in detail design to ensure adequate accessibility (12).

6. New detail design elements and trend in structure development

The development of advanced military aircraft does not only lead to application of new materials but also to the use of new detail design components the application of which if not thoroughly studied might involve some risk for the development program.
Prior to the use of new detail design elements, the design engineer therefore is compelled to request and conduct broad-based studies in order to eliminate taking any risk at all. Generally these studies should show trade-off character and have to be conducted along with corresponding structural tests to obtain a basis of sufficient broadness for evaluation purposes. As example several typical representative new components are dealt with in the following.

Here belongs for example the "safe-life" swing-wing bearing of an advanced strike aircraft shown in fig. 13 or the integrated thrust reverser of the same aircraft shown in fig. 14. Both parts are vital elements of the aircraft the failure of which might result in fatal consequences.

Aside from aerodynamic or thermodynamic studies, the form of design is mainly to be varified by early detail design investigations and functional and fatigue tests. The same applies to VSTOL aircraft where combined thrust deflectors / air brakes as shown in fig. 15 are required which in addition to aerodynamic loads have to take up high thermal stresses. In the same category of aircraft to be seen in fig. 16, airbleed control systems are used which are fed by high-pressure air from the engine system; they are subject to thermal and compressive stresses, occupy much volume for their installation and must be matched to given aircraft contours (13).

In consideration of increasing performance requirements and thus high sound pressure and heat loads, paying attention to excellent accessibility for good maintainability, and also for reasons of prolonged exposure to enemy threat and thus resulting vulnerability, the trend in development of structures for future fighter-type aircraft which possibly may have to follow a ballistic flight path, too, will be as indicated in the following.

The stressed structure and as shown in fig. 17 especially that of fuselages increasingly will be moved beneath the skin towards the interior where it is shielded from heat. In this way the fuel tank bays simultaneously will be located more inwardly, and the outer areas will be used for installation of subsystems, joining of heat loaded secondary structure or attachment of doors and panels.

7. Conclusion and recommendation

Fpr optimization of a weapon system and in particular realization of an aircraft structure in future the design engineer to a larger extent should be provided with

a) better inputs from aerodynamics and flight mechanics in nondimensional form, if not determined by tests then from a wide data bank,

b) quick and accurate methods for load criteria prediction and methods to analyse their impact on the aircraft configuration,

c) improved methods for weight prediction and weight control including retroaction on costs and flight performances,

d) aircraft lay-out programs with sufficiently spread weight subprograms to enable determination of basic configuration and growth factors,

e) clearly defined maintenance criteria and models,

f) reliability criteria,

g) models of enemy threat to be considered,

h) better information about new materials not only with respect to the static properties but with respect to fatigue strength and the corresponding information concerning corrosion, ductility etc.

i) clear information on available construction methods and

k) general instructions as to produceability.

It is necessary to computerize these information as far as possible so that they are at the disposal of the design engineer on his call.

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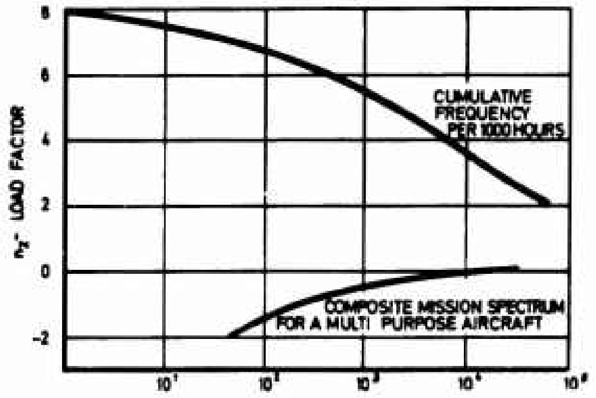


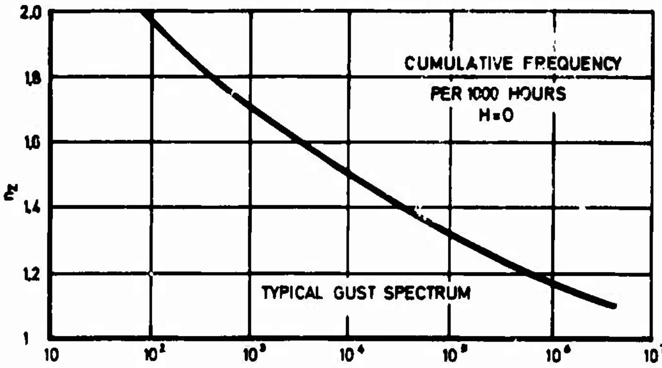
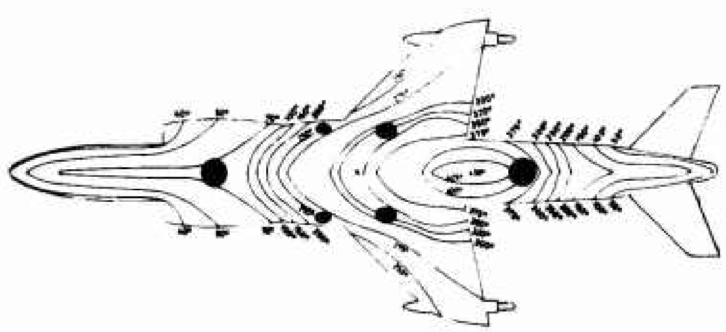


FIG. 1 TYPICAL MANOEUVRE AND GUST SPECTRUM



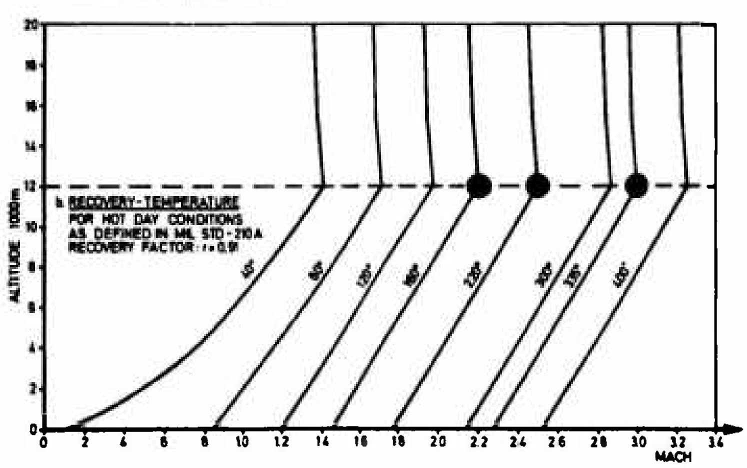


FIG. 2 TEMPERATURE DISTRIBUTION IN HIGH SPEED AND VTOL OPERATION

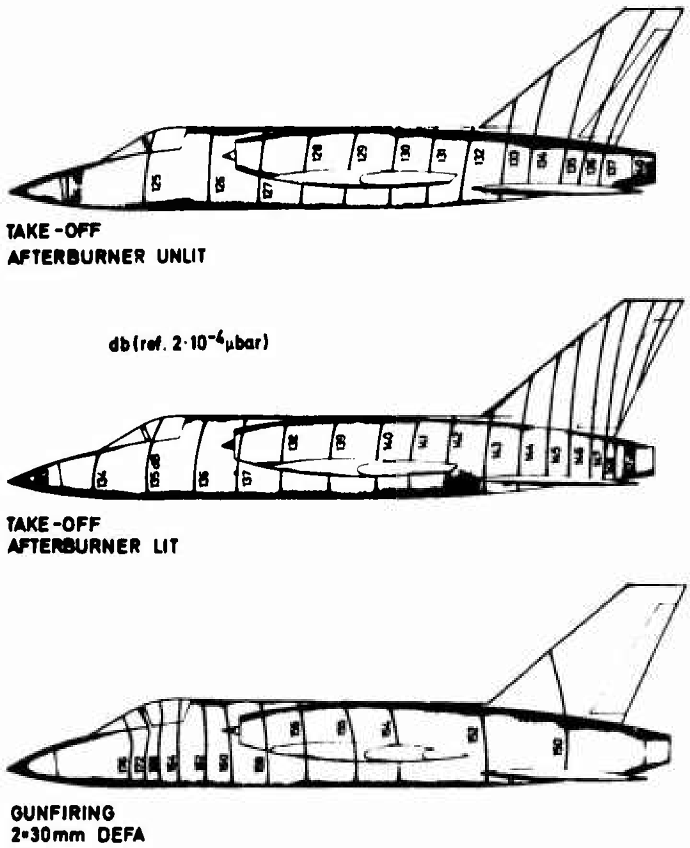


FIG. 3 SOUND DISTRIBUTION STOL A/C

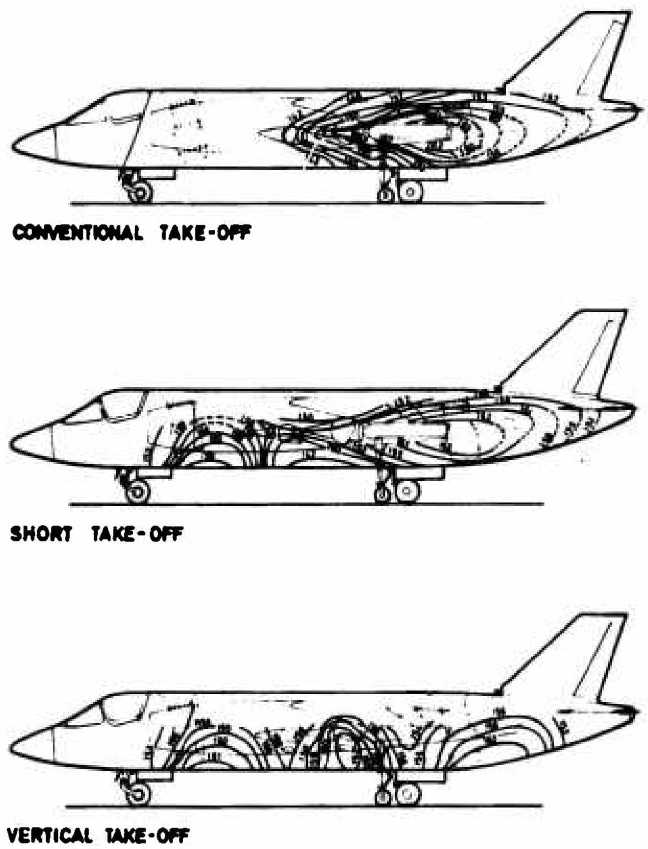


FIG. 4 SOUND DISTRIBUTION VSTOL A/C
VSTOL EXPERIMENTAL A/C VAK 191 B

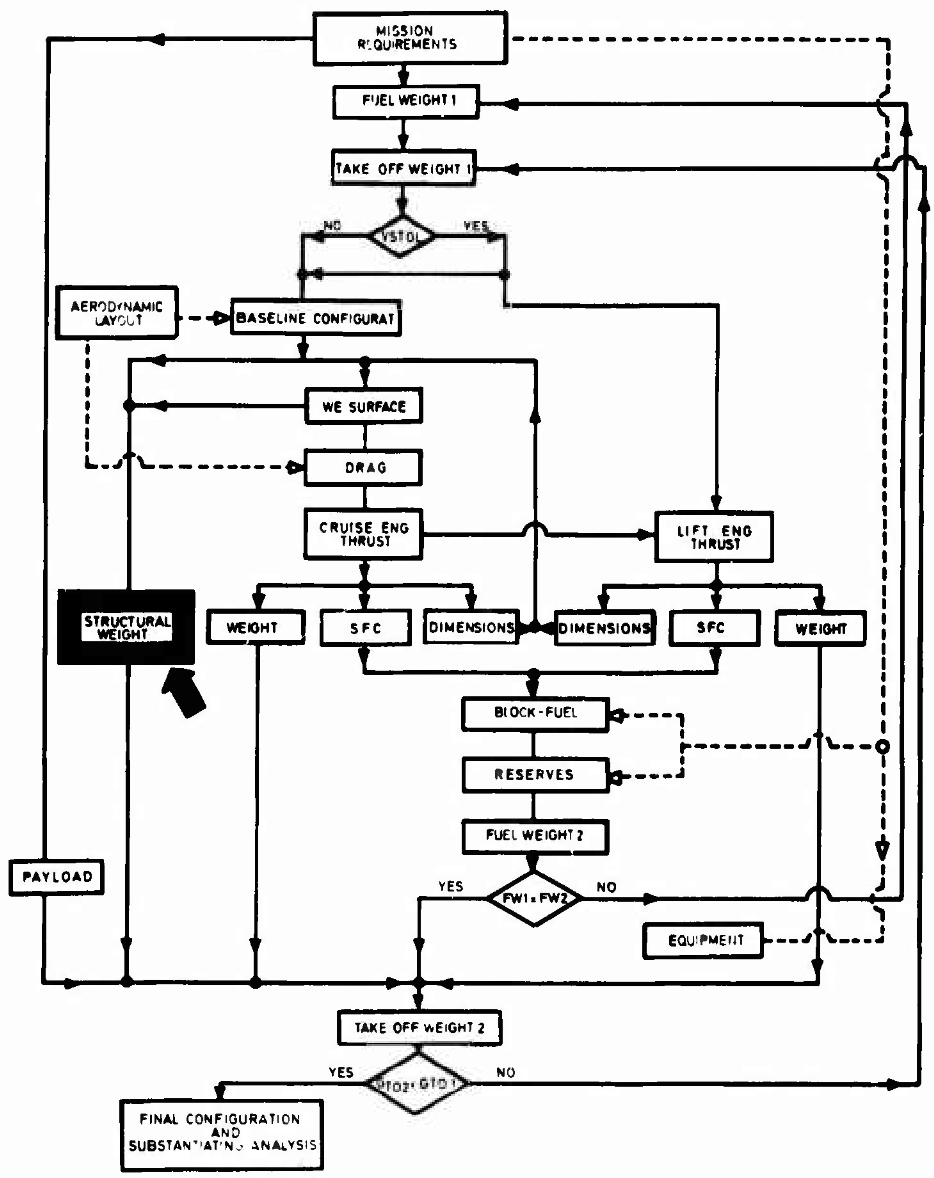


FIG. 5 AIRCRAFT LAY-OUT FLOW CHART

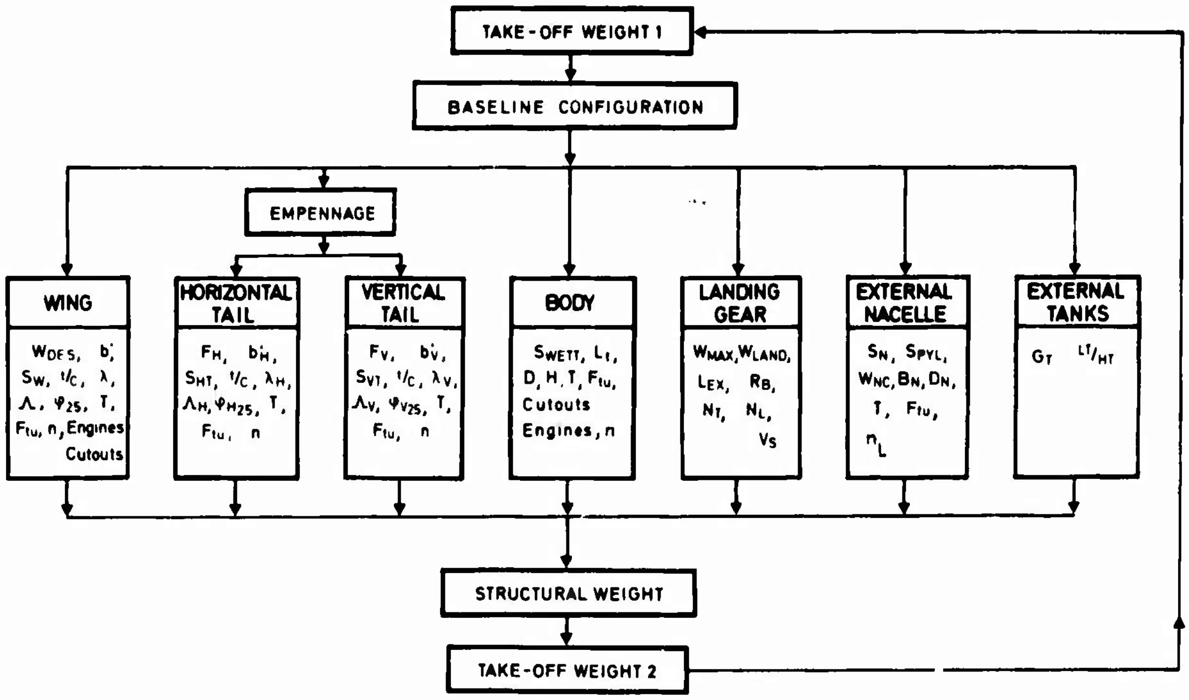


FIG. 6 STRUCTURAL WEIGHT FLOW CHART

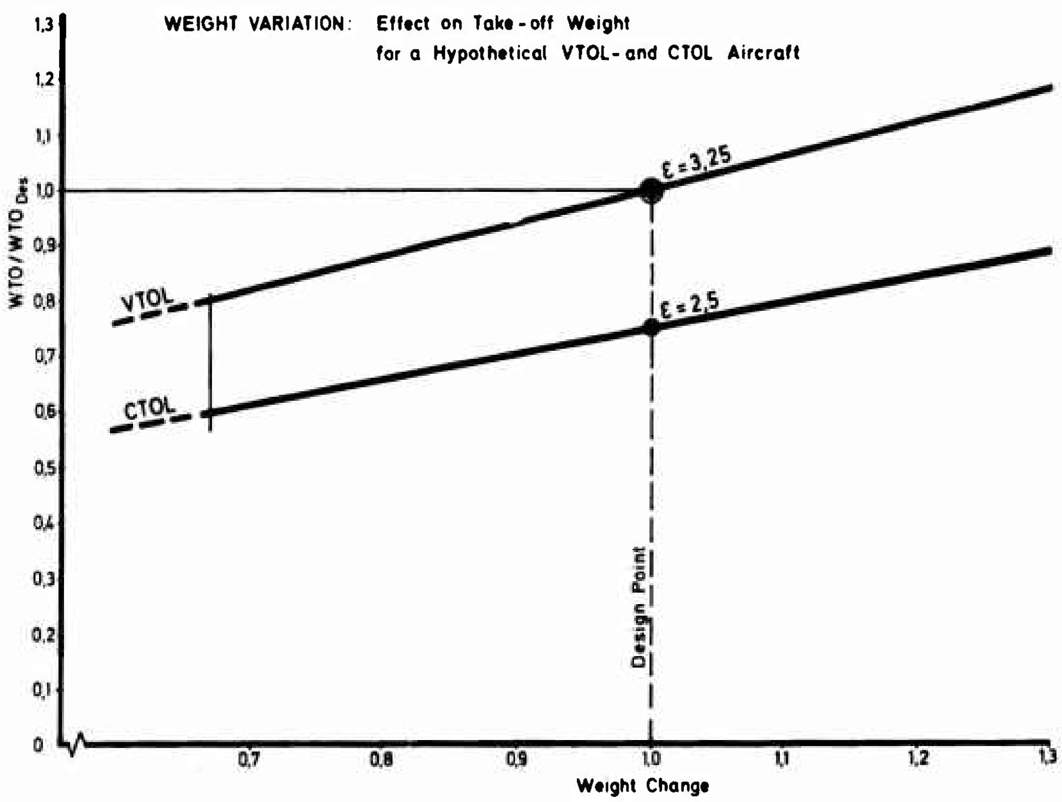


FIG. 7 AIRCRAFT GROWTH FACTORS

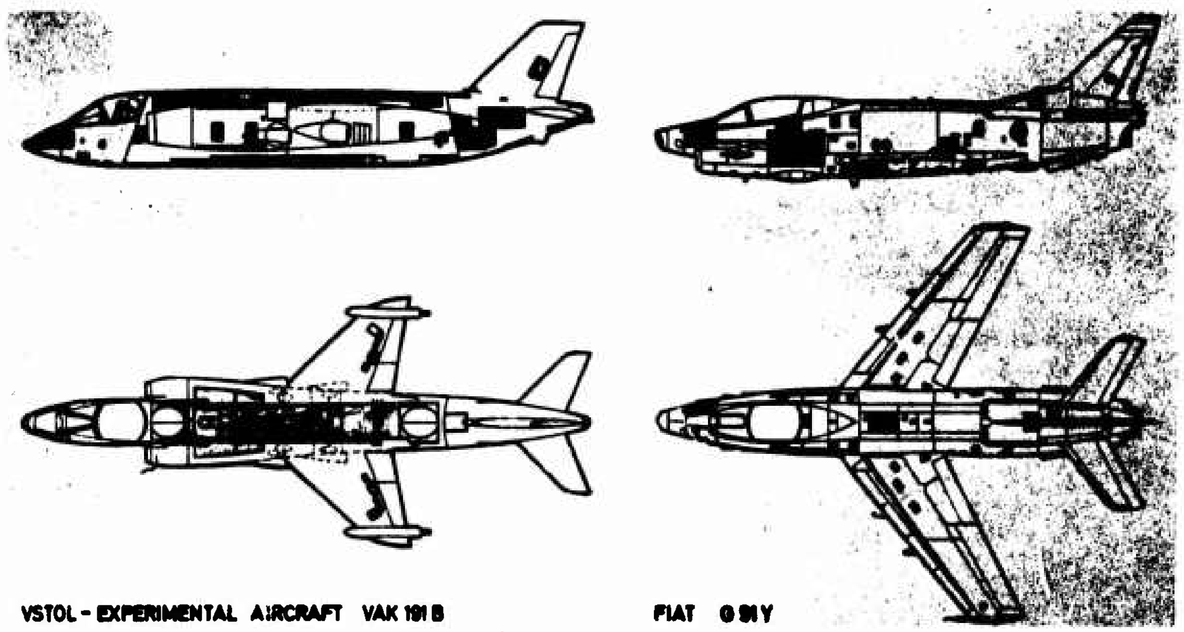


FIG. 8 MAINTENANCE AND ACCESS DOORS

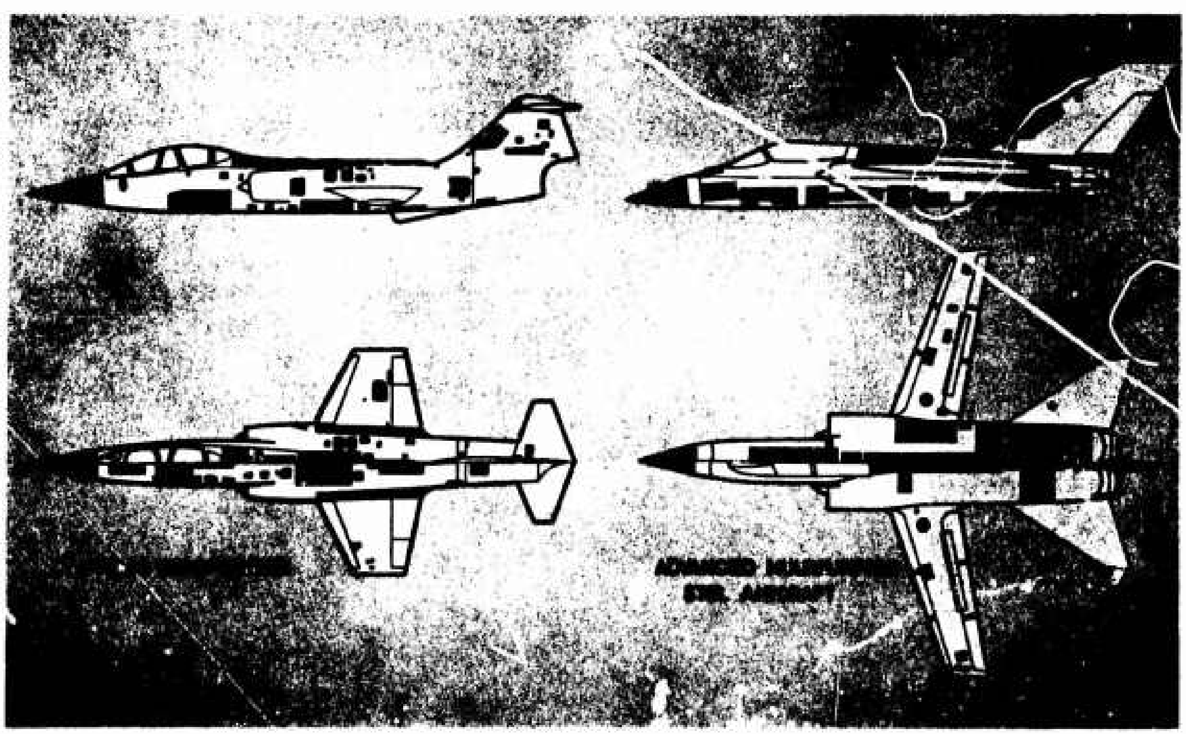


FIG. 9 MAINTENANCE AND ACCESS DOORS

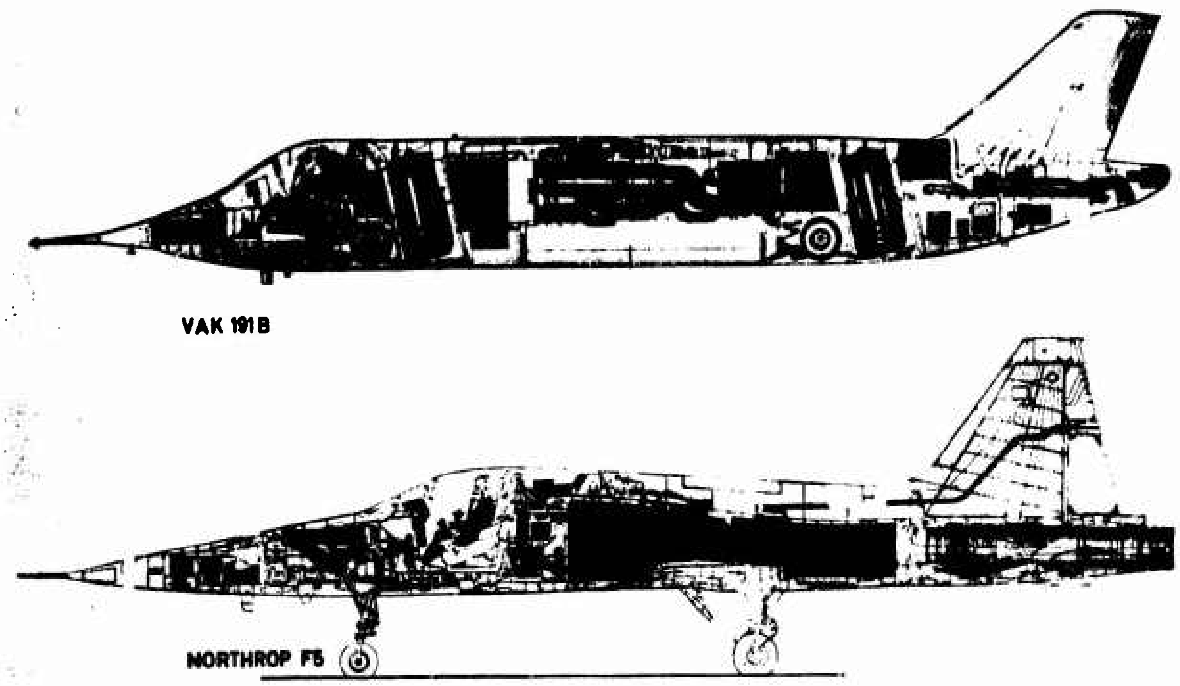


FIG. 10 EXAMPLES TO SHOW THE EQUIPMENT INSTALLATION DENSITY

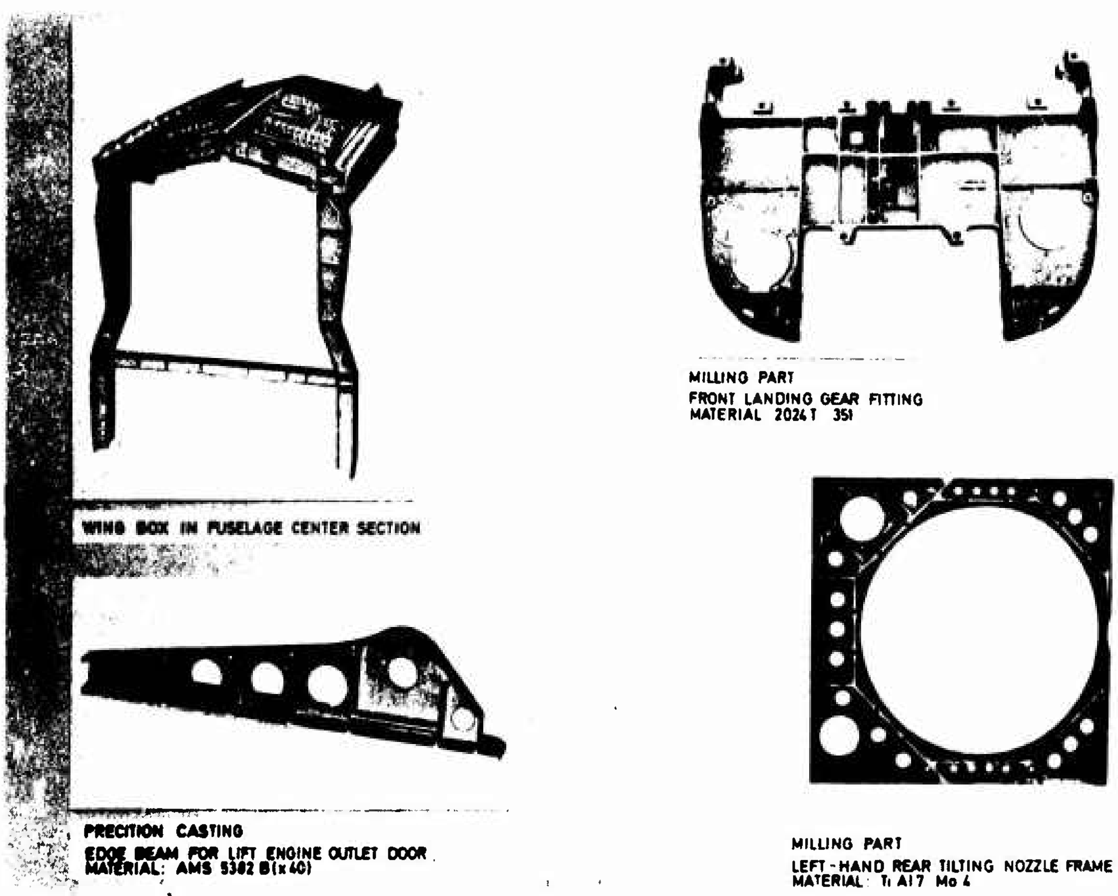


FIG. 11 INTEGRAL STRUCTURAL COMPONENTS

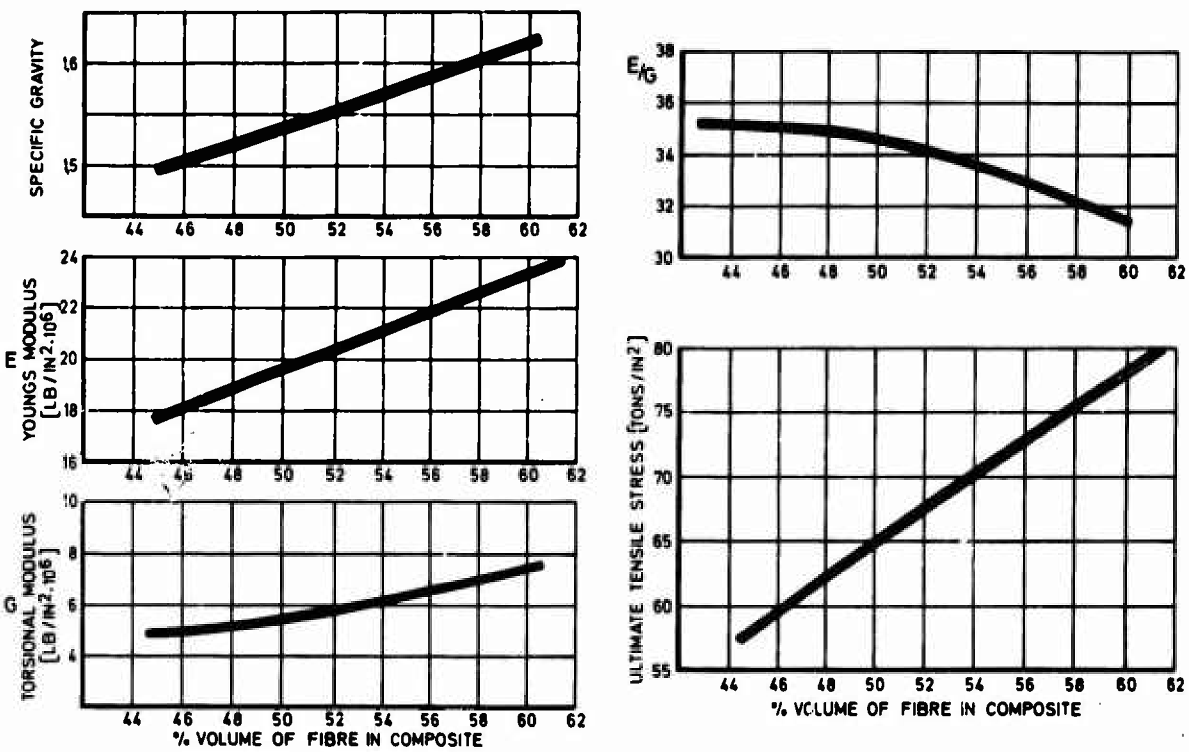


FIG. 12 PROPERTIES OF CARBON REINFORCED PLASTICS

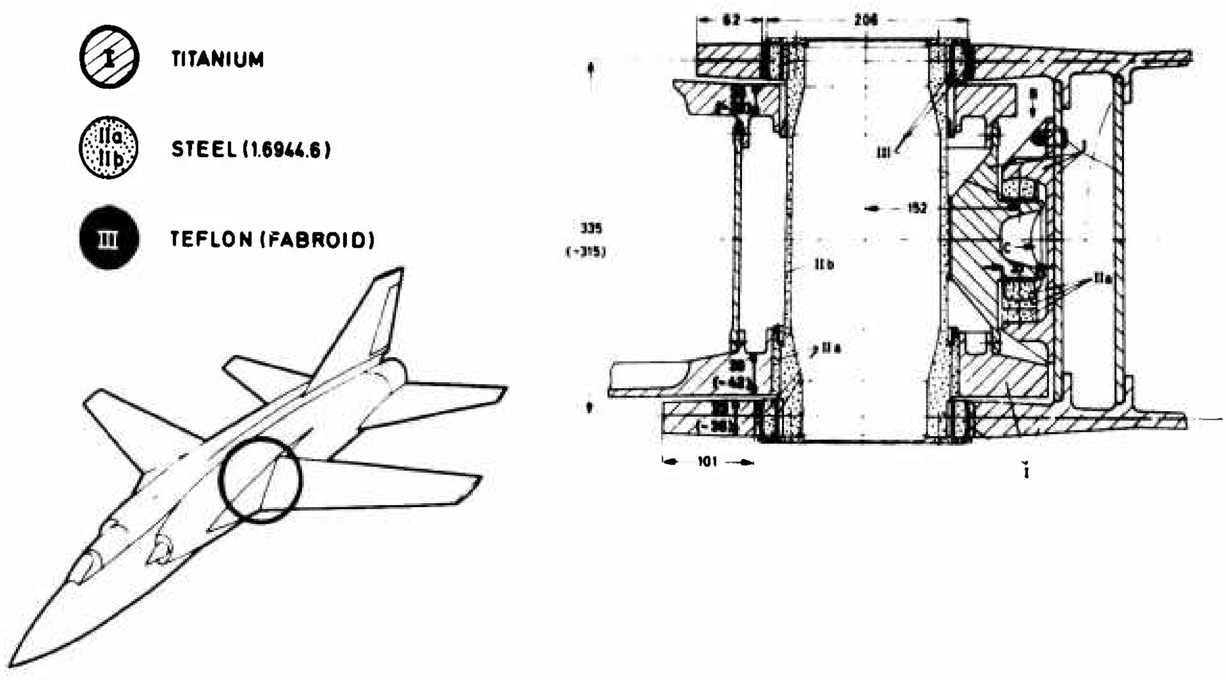


FIG. 13 WING PIVOT BEARING FOR ADVANCED FIGHTER

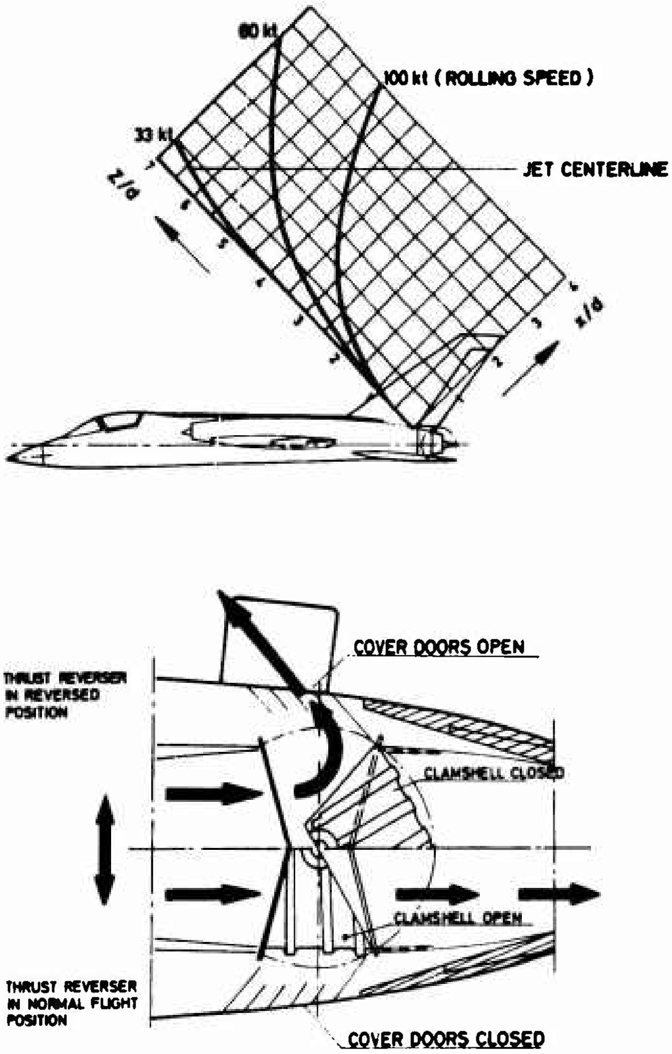


FIG. 14 INTEGRATED THRUST REVERSER

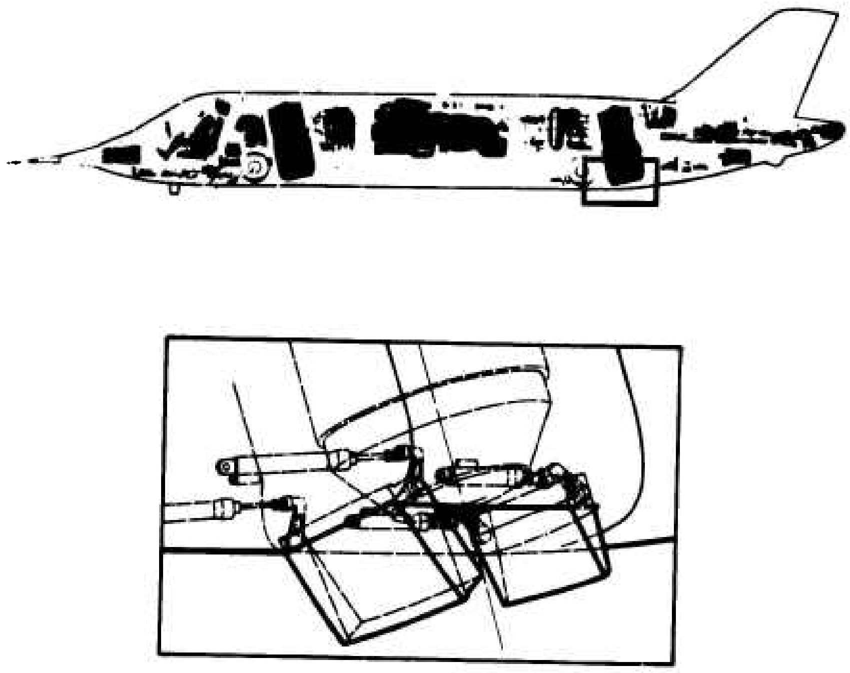


FIG. 15 COMBINED THRUST DEFLECTOR / AIRBRAKE SYSTEM

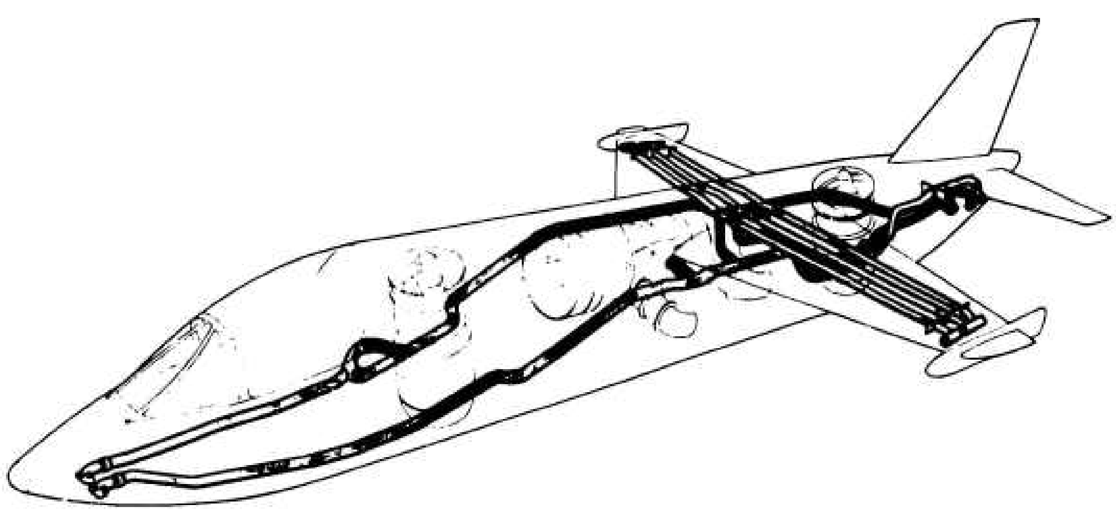


FIG. 16 VSTOL AIRBLEED CONTROL SYSTEM

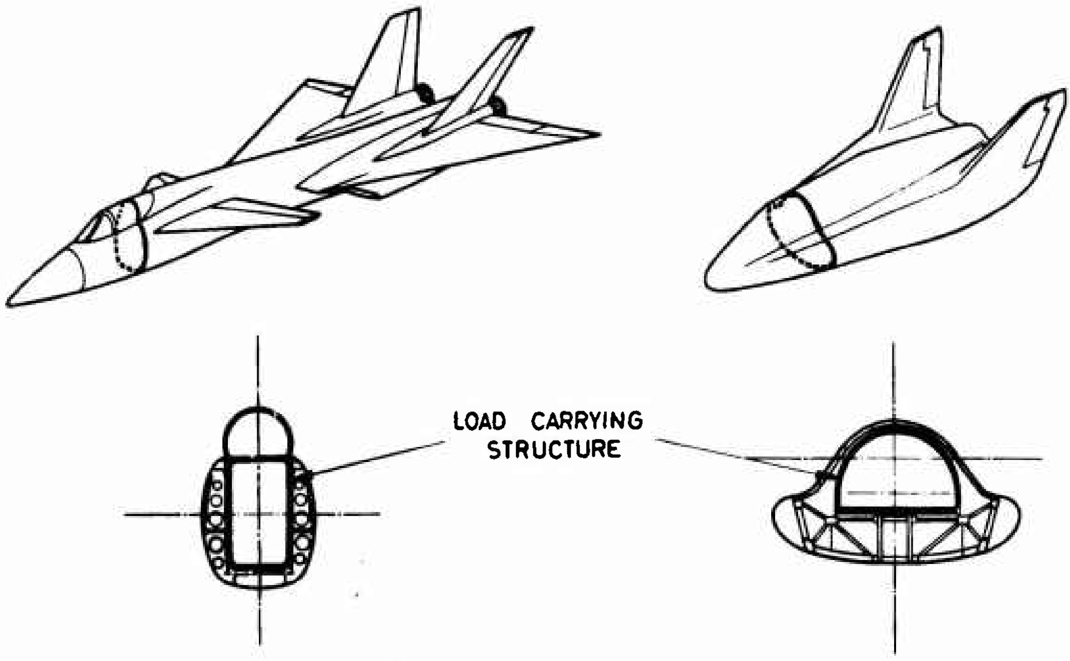


FIG. 17 FUTURE STRUCTURAL TREND

LOAD ESTIMATION AND AEROELASTICITY IN THE INITIAL STAGES OF ADVANCED COMBAT AIRCRAFT DESIGN

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S U M M A R Y

Within the areas of load estimation and aeroelasticity, the essential needs during the initial design stages of an aircraft are the rapid provision of realistic loading information and the exertion of a favourable aeroelastic influence on the design. This paper examines these needs for the tasks of load estimation and aeroelasticity in terms of three basic steps, namely, data acquisition, aircraft stability assessment and response calculations, with special reference to the problems posed by typical advanced combat aircraft configurations and requirements. The design of such aircraft highlights the need for an approach which integrates the various aerodynamic and structural disciplines, with the major problem being the acquisition of aerodynamic data, particularly for loading purposes.

1. INTRODUCTION

The tasks of load estimation and aeroelasticity are not fundamentally different for a particular type of aircraft, but the design of an advanced combat aircraft does emphasise certain features of the work. This emphasis stems either directly from the operational requirements, typical for such an aircraft, or indirectly from the type of configuration which results from satisfying these requirements.

The circumstances which prevail during the initial design stages of an aircraft also affect the nature of the work undertaken. There is seldom a predictable pattern of events during this evolutionary period: the elapsed time and the changes, both in configuration and requirement, can vary tremendously from one project to another.

Thus, before discussing the technical needs for the load estimation and aeroelastic tasks on an advanced combat aircraft, this report first attempts to identify certain objectives for the initial design stages. The techniques and resources, available for the fulfilment of both these aspects in conjunction, are then reviewed and the important deficiencies highlighted.

2. LOAD ESTIMATION AND AEROELASTICITY DURING THE INITIAL DESIGN STAGES

2.1 General Objectives

The primary objective, which confronts the loading engineer during the initial design stages, is the rapid provision of realistic quantitative quasi-static loading information, keeping in step with configurational changes. There is, however, a further and more positive objective which is often overlooked. This is the minimisation of the loads on the major components of the aircraft. The fulfilment of this objective is generally consistent with designing for good stability and control characteristics. Thus, if large loads are predicted within the required flight envelope, or rapidly increasing loads are predicted on the boundaries of the envelope, then alleviation by configurational changes should be investigated, as an alternative to increased structural strength.

The objectives confronting the aeroelastician are the rapid provision of quantitative quasi-static aeroelastic information (suitable for both loading and stability and control purposes) and assessment of the aircraft's flutter characteristics and any significant associated dynamic problems. Stemming from these aeroelastic investigations will come recommendations for stiffness, mass balance, damping or choice of configuration. The latter recommendation will cover items such as planform shape, store position and sensor [illegible]ing, for any automatic flying control systems. The primary aim of the aeroelastician [illegible], therefore, to exert a favourable aeroelastic influence on the design, in reconciliat[illegible] with other design considerations. Early involvment in the design cycle is the most effective way of achieving this objective.

2.2 The Pattern of Work

In the initial design stages, it is obviously impossible to supply full information on the loads or aeroelastic characteristics for every current configuration. To fulfil the broad objectives, it thus becomes necessary to identify or impose a pattern for any work undertaken, before specific tasks can be defined. Two classes of configuration are therefore recognised, the short duration configuration and the long duration configuration. At the outset, any configurational change must obviously be regarded as being of short duration, although in the event, it may prove to be a long one. On other occasions, the duration of a particular configuration may be deliberately sustained, even though it is not current. By this means the more detailed descriptions of a project, which are associated with a long duration configuration, can be obtained at suitable intervals. This provides the necessary basis for work in areas, such as load estimation and aeroelasticity, which demand fairly comprehensive data in order to proceed. A further distinction in the pattern of work can be drawn between the quantitative items of loading and aeroelastics, which define the quasi-rigid aircraft characteristics and design the structure, and the more qualitative nature of certain flutter and dynamic problem investigations. Whatever the duration of the configuration, some standard of information will normally be required for all the quantitative items. For the qualitative items, investigations are limited to significant items and significant configurational changes only.

2.3 Specific Tasks

For the long duration configuration, the aim is to supply detailed quasi-static loading information for all the major components of the aircraft, in a form appropriate to the method of structural analysis used. On the minor components only overall loading information i.e. total load and centre of pressure position, will normally be provided. Local pressures will also be estimated as required. Detailed calculations of the quasi-static aeroelastic effects are made for the major components, together with estimates of any significant

effects on the minor components. Flutter investigations on the long duration configuration, aim to make an initial assessment of all the primary surfaces independently. This is then followed by an assessment of any structural or aerodynamic coupling problems adjudged to be significant. Secondary surfaces e.g. flaps, rudders etc. may only be assessed in terms of simple criteria, rather than from comprehensive flutter calculations. When it is considered necessary, the determination of dynamic loads and vibration environment is also associated with the long duration configuration. However, such tasks will not normally be undertaken during the initial design stage, unless the proposed design has features which recommend investigation. Even then, the data produced should only be regarded as qualitative, since these problems tend to be sensitive to relatively minor differences in dynamic representation. A similar attitude is adopted towards any detailed assessment of coupling problems involving the basic aircraft and automatic flying control systems. Some detailed flutter and quasi-static aeroelastic trend studies may also be made for possible configurational variants, appropriate to the particular project; these studies essentially fall within the long duration category as well.

For the short duration configuration only overall load information for the major components is possible within a typical timescale. Estimates of quasi-static aeroelastic effects are made, but detail calculations will not be performed. Some estimates may also be made for other aspects of the aeroelastic task but, in general, no more than critical comment is provided. The basis of these estimates or comments is experience, supported by the results of prior long duration calculations.

3. CHARACTERISTIC FEATURES OF AN ADVANCED COMBAT AIRCRAFT

3.1 Operational Requirement

The starting point of any project is an operational requirement and the requirement for an advanced combat aircraft is typified by the following key features.

(1) An aircraft of reasonable size and weight.

(2) The ability to carry a large weight of stores selected from a wide range.

(3) Good airfield performance.

(4) High manoeuvrability (both symmetric and antisymmetric) particularly at high subsonic/transonic speeds.

(5) At least some supersonic performance.

(6) Good high subsonic low-level capability.

(7) Good ferry range.

3.2 Configuration

The satisfaction of such a requirement will, in turn, tend to produce certain characteristic features of aircraft configuration. Thus items 1 and 2 will almost certainly ensure an aircraft with external stores, mounted on wings and fuselage; the size of many of the store configurations will be large relative to the aircraft. Engines will need to be situated in the fuselage to leave the wings clear for store carriage. Item 5 is best satisfied by the use of swept, reasonably low aspect ratio aerodynamic surfaces, with a strong bias towards thin aerofoil sections, and fairly slender fuselage lines. The choice of wing, involves items 3, 4 and 7, which are suited by a low wing-loading and item 6, which demands a high wing-loading. The resultant compromise may produce a swept wing, of moderate aspect ratio and fairly high wing-loading, or the alternative solution of variable geometry. The wing will then have extensive high-lift devices, such as flaps and slats to meet item 3.

Pitch control will be achieved with an all-moving tailplane to cater for item 4 in conjunction with item 5. Since the needs of items 2, 4 and 5 will entail incorporation of a large effective fin, an effective yaw control - probably a large rudder - is then necessary to satisfy the cross-wind requirements, which will be a part of item 3. Roll control is best served by a combination of spoilers and differential tailplane, to cover the complete speed range. The choice of spoilers leaves the wing trailing-edge free for the incorporation of high-lift devices and gives a natural advantage of much enhanced rolling power, when these devices are in operation.

There are obviously other configurational features which are influenced by the operational requirement, but only those considered relevant to the subject matter of this report, have been summarised above.

4. TECHNICAL ASPECTS OF THE LOAD ESTIMATION AND AEROELASTIC TASKS

One major point, which is common to all technical aspects of advanced combat aircraft design, is the size of the task. This arises from the large number of configurational variants possible from meeting store carriage requirements and the significant changes to the clean aircraft mass and aerodynamic properties, produced by these stores. If variable geometry is embodied in the design, this further increases the number of configurational variants to be considered. The volume of work associated with tasks involving a high aerodynamic content, such as load estimation and aeroelasticity, is also influenced by the need to consider subsonic, transonic and supersonic flight conditions.

Now the typical configuration characteristic of thin aerofoil sections on an aircraft operating at high dynamic pressures inevitably demands consideration of aeroelastic phenomena. The carriage of stores is also a potential source of dynamic problems. Nevertheless, the size, weight and stiffness characteristics of the typical advanced combat aircraft should provide sufficient frequency separation between the rigid aircraft motions and the elastic modes of the structure, to permit the usual practices of considering the quasi-rigid and elastic representations of the aircraft independently. These quasi-static aeroelastic effects need to be determined, not only for stability and control investigations, but also for loading purposes. Temperature effects, on the stiffness properties, due to kinetic heating, will be minimal for the typical supersonic performance requirements.

The design loads are controlled by the requirements of such documents as Mil. Specs. or AvP970, and the form of these requirements means that the majority of design conditions are associated with the quasi-rigid aeroplane. The high manoeuvrability requirement for a combat aircraft also means that these design loads are predominantly based on manoeuvre cases. The cases can be provided by either symmetric or asymmetric manoeuvres: true antisymmetric manoeuvres will seldom occur because roll-coupling tendencies are inevitable for the typical configuration. Lateral controls, such as spoilers, contribute to the degree of asymmetry present in such manoeuvres, since they produce asymmetric inputs. Rudders and differential tailplanes, although nominally yaw and roll controls, will produce significant inputs to all the lateral degrees of freedom of the quasi-rigid aeroplane. To some extent, a favourable influence on roll-coupling behaviour can be exerted by the use of coupled control gearings, but it is not really practicable to cater for all regions of the flight envelope.

The major implication of roll-coupling behaviour on the loading task is that it is no longer possible to define loading cases simply, on the basis of limited degree of freedom responses e.g. rolling. It then becomes difficult to identify possible critical flight conditions and configurations and, for certain parts of the aircraft, extremely difficult to associate the critical loading condition with the occurrence of a significant event in the time-history of the response. In these circumstances, a large number of responses need to be examined, with monitoring of significant stressing parameters. The definition of a significant stressing parameter can, in itself, prove difficult for typical advanced combat aircraft structures. Taken to the ultimate, this process could involve the monitoring of individual stresses, but it is obviously preferable if parameters, such as, bending moment or combinations of bending moment, shear and torque can be used.

An outcome of high wing-loading design in conjunction with the high manoeuvrability requirement, is to produce design loads for the major components which are associated with high aircraft incidences at high subsonic/transonic Mach Numbers. The design loads therefore need to be defined for flight conditions where the aerodynamic characteristics are decidedly non-linear and sensitive to small changes in Mach Number and incidence. The carriage of underwing stores can also have a profound effect on these aerodynamic characteristics. The corresponding quasi-static aeroelastic effects therefore, need to be defined for a non-linear system of equations governing the motion of the quasi-rigid aeroplane. If the design features variable geometry, this need to define aeroelastic effects for a non-linear system, extends to lower incidences, when the wing is fully swept.

Further aeroelastic phenomena, associated with a variable geometry configuration, arise from wing-tailplane aerodynamic interference effects. If the geometric separation between the wing and tailplane is small, quasi-static aeroelastic distortions essentially change the geometry of the aircraft and thereby introduce another source of non-linearity, via the interference effects. Wing-tailplane aerodynamic interference effects must also be considered in assessing the flutter characteristics of the aircraft. On advanced combat aircraft, in general, flutter considerations are most likely to have design repercussions on the tailplane and the wing with stores. The latter may also produce dynamic loading problems due to carriage (both in the air and on the ground) and when they are jettisoned. Quasi-static aeroelastic considerations will normally influence the fin and rudder design and thereby ensure freedom from flutter. The size of the typical aircraft and the high design loads, make flutter and dynamic problems associated with the skin panels, very unlikely.

For dynamic loading problems involving responses of a transient, rather than an oscillatory nature, difficulties can occur, similar to those previously mentioned in connection with quasi-static loads from asymmetric manoeuvres. These are the difficulties of identifying critical cases and of specifying significant stressing parameters, both of which can become very difficult if a large number of elastic modes is needed for adequate problem representation.

5. THE TECHNIQUES AVAILABLE FOR LOAD ESTIMATION AND AEROELASTICITY

5.1 Fundamental Description of the Tasks

Discussion of the available techniques is assisted by describing the two tasks in the more fundamental terms given by breaking down the necessary processes into three basic steps, namely,

(i) Data Acquisition

(ii) Stability Assessment

(iii) Response

The first of these steps is concerned with the acquisition of all detailed mass, elastic (structure and systems), flying control systems and aerodynamic information, necessary to represent the aircraft. This representation will include automatic flying control systems when they are appropriate to a particular study. The second step involves assessment of the stability of the overall aircraft and the final step is broadly concerned with the response of that aircraft to various types of input. As previously mentioned, the latter two steps will normally be subdivided into quasi-rigid and elastic investigations. It should also be noted that, throughout this report, definitions of stability are confined to problems described by equations of a linear constant coefficient type. For other types of equation, any assessments made are described in terms of the response to a defined input.

For quasi-rigid investigations, if the incremental aerodynamic and elastic forces are linear functions of the distortion, it is possible to perform the customary intermediate stage of introducing aeroelastic corrections to the coefficients of the rigid equations. The equations which govern this intermediate stage are then in the form of a response problem, with aeroelastic distortion as the unknown variable and the forces due to rigid aircraft motions as the input. Since dynamic pressure may be written as a scalar operating on the matrix of incremental aerodynamic forces, the stability roots of this system of equations can also be determined, in terms of critical values of dynamic pressure. An unstable real root indicates a static divergence, but, in general, real roots of either sign and complex roots will exist. Response solutions, for a specified value of dynamic pressure, give the corresponding aeroelastic distortions and hence the aeroelastic corrections. If the incremental aerodynamic forces are non-linear functions of distortion, then the above stability concept is no longer valid and if these forces are also functions of the rigid aircraft motions, then aeroelastic corrections can no longer be applied. Solution of the quasi-rigid response problem, in the latter circumstances, requires the equations of motion of the rigid aeroplane to be considered inclusive with the equations of aeroelastic distortion.

Describing the tasks of load estimation and aeroelasticity in terms of stability and response, automatically involves the 'stability and control' aspects of aircraft design. This is inevitable, because although the stability of the quasi-rigid aeroplane might possibly be regarded as a distinct task, the response aspects are indistinguishable, whether the output be quasi-static loading data or output for handling criteria. In this context, it is worth noting that even when aeroelastic corrections are permissible, some terms in the differential equations, governing the quasi-rigid motions of an advanced combat aircraft, will not normally be of a linear constant coefficient type. Stability assessment is then replaced by an assessment of the response to suitable pilot inputs. The stability and response of the elastic aeroplane have more usually been treated as related topics.

5.2 Data Acquisition

5.2.1 Mass, Elastic and Systems Data

The last decade or so has seen the development of matrix methods of structural analysis (in parallel with the development of the digital computer) throughout most of the aircraft industry. These methods are almost universally applicable, regardless of configuration, so that a theoretical definition of an aircraft's elastic properties is always possible, although the preparation of input data can still be a considerable task. When they are applicable, the more traditional, simple beam methods of structural analysis may be used instead, but these methods are not adequate for many structural components typical of an advanced combat aircraft. However, structural idealisation is the precursor to any form of structural analysis and this process is not so amenable to precise theoretical treatment. Many simplifying assumptions must be made at the idealisation stage and resort to data sheet methods may also be necessary, particularly for many of the structural attachments between components. Nevertheless, satisfactory methods of idealisation have been developed for many structural items and the publication of structural data sheets is well established. To the latter can be added unpublished information obtained by an individual firm, from test specimens made for the specific project or a prior one.

Use of the above theoretical methods of analysis, presumes a linear relationship between forces and displacements for the aircraft structure. This is the usual assumption made in defining the elastic properties, although variations in stiffness due to the changing geometry of mechanical linkages are considered. The assumption of linearity is probably reasonable in most circumstances, but there are two possible exceptions: these are when the structure is lightly or heavily loaded. In the former case, back-lash or soft-centring may exist, whilst in the latter, panel buckling may occur. A further deficiency of the theoretical methods is the inability to predict structural damping properties, though the order of typical values is well known for the normal modes of vibration measured during resonance tests on existing aircraft.

Systems data fall into two distinct categories. Data within the first category originate from investigations made for the purposes of specifying the required system properties. These data are predominantly theoretical and will be produced by the aircraft design organisation: the data will not normally provide a system description which is completely adequate. Data within the second category are based on the actual system properties, as supplied by the equipment manufacturers, and are only known when the hardware has been defined. Unless existing hardware, or something similar, are incorporated in the design, then data within the second category may not be defined until late in the design cycle, as information becomes available from bread board studies (when appropriate) or hardware testing. At different stages throughout the design period, various combinations of data from theoretical representations and experimental analyses will be used. The definition of system characteristics suitable for loading and aeroelastic purposes will often involve some prior response investigations of a complex system; this system may be a single item or a combination of several. Such investigations will normally require computational aids and both digital and analogue computers are used. The latter are often favoured, because of the built-in non-linearities which are typical of many practical systems, but for many analyses some degree of linearisation is accepted.

The provision of adequate detailed mass information has generally been a possible, even if a somewhat laborious task. Compared to simple beam methods of structural analysis, requiring mass, moment and moment of inertia properties at a limited number of stations, the advent of matrix methods of structural analysis has considerably increased the magnitude of this task. Thus for an advanced combat aircraft, masses will generally need to be defined at the node points of structural analysis grids, for most of the aircraft components.

Apart from the difficulties sometimes encountered in obtaining satisfactory systems mass information, the major outstanding problem of mass definition concerns the mass (or more strictly the dynamic properties) of fuel for other than steady-state conditions. Some limited theoretical and experimental information on fuel-sloshing behaviour exists, but mainly for regular tank shapes which are not characteristic of most aircraft fuel tanks.

Before aeroelastic work can proceed the representation of the aircraft needs to be of a manageable size, so that a reduced form of description of the aircraft's elastic properties is normally required, particularly if matrix methods of structural analysis have been used. This is best achieved, firstly, by describing the aircraft in terms of branch items; these items will correspond to the major components of the aircraft and any other components which have independent elastic characteristics. Parameters which are aeroelastically significant, such as control actuator stiffness, are included in the latter category. The next stage of reduction is to represent the structural components by a minimum, but sufficient, number of displacement and angular freedoms. For flutter and dynamic problems, reduction is then carried a stage further and the dynamic characteristics of these components are represented by a limited number of normal modes of vibration. This involves the solution of eigen value problems with real roots and, as such, presents no difficulties for quite large order matrices on a digital computer. Reassembly of the component modes and the branch items will then give the required reduced description of the total aircraft.

One other essential requirement for aeroelastic work is the ability to express the displacements and angles at points on the aircraft, relevant to the aerodynamic description, in terms of the displacements and angles given by the structural analysis. This is not an exact process, so that various methods, from graphical interpolation to digital programs for polynomial and spline fitting techniques, have been evolved over the years. The difficulty of defining a suitable transformation process is much greater when matrix methods of structural analysis have been employed and none of the existing published methods is entirely satisfactory.

5.2.2 Aerodynamic Data - Classification and Overall Needs

The major difference between the aerodynamic data and the other types of data required, is the strictly limited capability of current theoretical aerodynamic methods. Aerodynamic data are therefore obtained from three different sources: these sources are:-

(i) Theoretical Data

(ii) Measured Data - from wind-tunnel measurements directly appropriate to the particular design configuration.

(iii) Empirical Data - correlated data from data sheets or measurements on like configurations.

The above definition of measured data means that if the project configuration changes from that tested, then the corresponding measured data are reclassified as empirical data. These data sources are discussed in more detail in Section 6.

The required aerodynamic description of the aircraft is based on an amalgam of data from these three sources. For loading and aeroelastic work, this description needs to be defined in terms of forces (and moments if required) at points relevant to the structural description. Thus, forces will be defined at the grid node points for matrix methods of analysis and bending moments, shears and torques at various spanwise stations for simple beam methods. It is also preferable that consistency is maintained between these quantities and any pressure information. Exact methods for these processes do not exist and most sections of the industry will have proprietary methods, with the more precise methods demanding digital computation. These processes can be avoided wish for the customary approach to flutter problems and are sometimes avoided for quasi-static aeroelastic calculations, but if loading information is subsequently required, this is a short-sighted approach.

5.3 Stability Assessment

Before the advent of computers, both analogue and digital, the only difficulty in obtaining stability solutions was one of computational effort, with a three degree of freedom, second order system, providing a practical limit. These days it is a fair generalisation to state that the stability of both quasi-rigid and elastic representations of an aeroplane can be determined without difficulty, for an adequate number of degrees of freedom. This statement is also applicable to stability problems involving aeroelastic distortion, such as static divergence. Analogue computers still have some merits in flutter work for making rapid assessments of parameter changes, but the number of degrees of freedom is restricted. Stability solutions are therefore mainly obtained by using digital computers and, in conjunction with plotting facilities, these solutions are readily presented for easy assimilation.

For flutter problems involving many degrees of freedom there are, however, quite often difficulties of interpretation. Apart from the natural wish for basic understanding, interpretation of the results is necessary if cures are sought to a potential flutter hazard. It also helps to provide a basis for a minimum of controlled parameter changes, in assessments of problem sensitivity. Formulating the problem in terms of branch modes enables structural and mass parameter changes to be investigated easily, but it is of less help for studying aerodynamic variations. An approach (1) which reduces any flutter problem to binary form, promises a possible answer to the difficulties of problem interpretation and also minimises the number of parameter variations to be considered.

There is another method of assessing the stability of the elastic aeroplane, which obviates the need to define aerodynamic data, and that is to build wind-tunnel flutter models. These can vary tremendously both in complexity and scope, ranging from large low-speed complete aircraft models, with interchangeable components, to small high-speed models of perhaps a single surface. Tests can be made, either by trying to establish the critical flutter speed (employing measures aimed at preventing model destruction) or by using a sub-critical response technique to determine the roots of the system, stopping short of actual flutter.

5.4 Response

Since the response of a linear system, in essence, merely involves solution of the stability equations with appropriate forcing terms, many of the comments made on stability solutions in the previous sub-section also apply to the response. Thus the impact of computers on the solution of linear response problems is identical but, in addition, they have enabled larger order non-linear problems to be solved. This has been of particular benefit to the investigation of roll-coupling behaviour and to the definition of the related quasi-static loads. Digital computation is essential to solution of the quasi-rigid response problem with non-linear aerodynamic forces due to aeroelastic distortion.

Most of the responses investigated are for the determination of loading information from the response of the quasi-rigid aeroplane, to inputs defined by official design requirements. Either steady state solutions e.g. longitudinal balances, or transient solutions e.g. control inputs and discrete gusts, are required. However, in parallel with computer development (digital in particular), there has been an increasing interest in random and dynamic loading problems, which is now being reflected in some of the official design requirements. These problems tend to fall into two categories, namely, responses originating from either environmental or self-generated excitation. Examples of the former category are atmospheric turbulence and rough runways whilst store jettison, system failures and buffet fall within the latter category.

Obtaining solutions to these types of problem presents no additional difficulty, with either transient and indicial or oscillatory steady state solutions (for the preparation of power spectra, etc.) being required. Careful choice of method and the form of results presentation can, however, materially assist in the interpretation of the large order problem. An experimental approach to obtain solutions of buffet loading problems can also be adopted, utilising the dynamically scaled models produced for flutter testing. This again obviates the need for aerodynamic data acquisition, in particular, the difficulty of defining the input forces due to buffet. However, strength limitations restrict the level of steady-loading to which the models may be subjected and some reservations must be retained about this approach, because of possible scale effects.

6. THE ACQUISITION OF AERODYNAMIC DATA

The three sources of aerodynamic data have been identified in the previous section. This section enlarges on these data sources and reviews the means of producing an overall aerodynamic description of an aircraft.

6.1 Theoretical Data

Current three-dimensional aerodynamic theories (which are necessary to combat aircraft design) are strictly limited to potential flow assumptions on wings or slender bodies of revolution, in truly subsonic and supersonic flow conditions. Subsonically, a trailing edge flap theory has yet to be established. There are, however, linearised subsonic interference theories now available for two surfaces in parallel planes (2) (ignoring wake effects) and for T-tails without dihedral (3). Unsteady aerodynamic theories are strictly valid for only oscillatory motions and invalid for transient conditions. Various theoretical methods for the isolated wing are used throughout the industry, but reservations must be expressed about the assumptions made and the numerical integration techniques used, for some subsonic theories, whilst certain supersonic theories are not orientated towards the provision of loading and pressure distribution information. Digital computation is essential for all theoretical aerodynamic methods.

6.2 Measured Data

The commonest form of measured data available, comes from wind tunnel measurements of overall steady force and moment data on rigid models of the complete aircraft. These tests will normally cover different tailplane settings, for combined ranges of incidence and sideslip angle, over the required Mach Number range. Tests may also be made for other control surface settings such as differential tailplane, spoiler and possibly rudder. These latter control surface settings will be tested independently, for perhaps a couple of different tailplane angles and probably over a limited Mach Number range. Low-speed model tests will also be made for combined ranges of incidence and sideslip angle with various combinations of slat, flap and spoiler settings. All of the above tests are orientated towards stability and control work but for quasi-static loading purposes it is normal practice to cover the incidence and Mach Number range with the tailplane off. Occasionally some lateral testing may be made with the fin off. Loading information may also become available from strain gauging of flaps, slats, spoilers and, possibly, undercarriages and airbrakes on low speed models.

Models specifically designed for quasi-static loading purposes are pressure models and component load models. The latter type of model is strain-gauged to measure forces and moments on major aircraft components i.e. wings, tailplane etc. and may also include other components such as stores. (Strain-gauged stores may sometimes be fitted to the models measuring overall forces and moments, but this is a far less satisfactory arrangement). Component load models will normally be tested for the same range of parameters as the overall stability models but pressure models may be tested for a reduced number of sideslip angles and Mach Numbers. Further pressure information is sometimes available from other models, particularly low speed models, which have not been specifically built for loading purposes.

Unsteady measurements tend to be confined to a few special rigs or facilities designed for these purposes. Two main types of testing are made and both types are almost invariably for small amplitude oscillatory motions of nominally rigid complete, or component, models. In the first type of test, forces and/or moments are measured and in the second type, pressures are measured. Tests of the latter type are one of the more interesting recent developments (4). Unsteady tests will be made for one, or possibly two, amplitudes of oscillation and a minimum of incidence and sideslip conditions, at Mach Numbers of particular interest. For a complete model, oscillations in five rigid degrees of freedom are possible, whilst for a surface,

two, or three, displacement and angular freedoms (producing motions normal to the plane of the surface) will be available. A trailing edge flap, rudder etc., would be restricted to angular motions about the hinge.

In order to provide some guidance on quasi-static aeroelastic effects for the particular variable geometry problem of a highly swept wing, with a tailplane in close proximity, another type of model has recently been tested at the author's firm. For this model, a series of distorted wings was manufactured, with the same non-dimensional shape but different magnitudes of distortion. This shape was based on theoretical aeroelastic distortions, but only bending and root flexibilities were considered as the shape was found to be relatively insensitive to torsional flexibility. The theoretical calculations also showed that it was practicable to choose a unique shape, since only small differences in this shape were indicated over the incidence range. Overall and component measurements of force and moment were made for the range of parameters, usual for these types of model.

The increasingly large amount of data produced by wind-tunnel testing, particularly by component load and pressure plotting models, makes stringent demands on the processing capacity of an organisation. The digital computer has already made some impact on this task, in providing processed graphical and numerical data. On occasions these data may also be made available on a magnetic tape, either for subsequent operations, such as pressure integration, or as a limited form of data bank, but there is still much that can be done in this field.

6.3 Empirical Data

Empirical data sources cannot be summarised by quoting a few well established documents, since such documents do not exist. Documents of the data sheet type e.g. R.Ae.S. data sheets and Datcom, can be of some use for determining quasi-static component loads, but they are really preoccupied with providing rigid aircraft data for stability and control purposes. Thus, for example, one outcome of this exclusiveness is that spanwise centre of pressure positions are not normally considered, unless they are required in the estimation of a particular derivative. Also the reliability of some of the data sheet methods becomes suspect if the flow is not attached, so that their utility for loading purposes is thereby depreciated. Apart from deficiencies of the above nature, the use of such data sheet methods for load estimation demands that plausible load or pressure distributions can be associated with the estimated component loads and their point of action. This will often place more stringent requirements on the accuracy of such estimates. Data sheet methods for the determination of the required load and pressure distributions are few. Those that do exist are usually based on two dimensional and lifting line or limited three dimensional theories, applicable to fairly high aspect ratio wings, flaps or trailing edge controls. Such methods do not contribute greatly to the processes of load estimation, though they may be relevant to unswept wing configurations on a variable geometry aircraft at moderate incidences.

There is obviously a vast amount of measured wind tunnel information, both published and unpublished, which gives overall aircraft steady forces and moments on rigid models of various configurations. There is somewhat less data on steady forces and moments for the major aircraft components and components such as flaps, slats, spoilers, trailing edge controls, airbrakes and stores. A still smaller quantity of data will exist in the form of measured steady pressure data. To all this data can be added the relatively small amount of aerodynamic information, suitable for loading purposes, available from flight measurement. Of this total accumulation of information, presumably some of it has influenced the existing data sheet methods for stability and control purposes, but the main interest for loading work concerns those items not covered adequately by existing data sheets, namely, the forces, moments, load distributions and pressures on components, particularly at high incidences. Unfortunately, little of the testing made to obtain data on certain of these items has been systematic, and collation and correlation of such data has not been attempted to any great extent. Most design organisations will have accumulated empirical data for their own use and some of the more valuable data will be that obtained en route to the current configuration of a project. However, the use of this data, like all empirical data, presumes the ability to correlate it in terms of significant parameters. Too often the methods of utilisation are born of expediency.

The amount of reliable published unsteady information provided by wind tunnel testing is fairly small, particularly from the relatively recent technique of unsteady pressure measurements. Forces and moments measured on rigid wing models subjected to small amplitude pitching and heaving motions (rolling also for half models) constitute the majority of available data and a good proportion of these data are for low reduced frequencies. Some of the more useful measurements concern quantities, such as wing pitching moments and trailing edge control hinge moments, which may change rapidly in both magnitude and/or sign in transonic regions. In general, however, there is not sufficient data from unsteady measurements to permit very effective correlation. One other source of empirical unsteady data remains and that is two dimensional theory. The classification of empirical is considered justified, because an aeroplane is indisputably three dimensional. The theory needs to be modified to give sensible lift slopes (usually, simple overall factors are applied) but it may then be used for flutter work at moderate reduced frequencies on surfaces of reasonable aspect ratio in subsonic flow conditions. Supersonically two dimensional methods are of little use for the typical advanced combat aircraft configuration, except possibly on flaps or trailing edge control surfaces.

Unless a firm has chosen to construct a digital source of data, published empirical data are normally available in either graphical or tabular form, but occasionally they are presented as empirical formulae. Unpublished data are often in a similar form, but if they have been obtained directly from wind-tunnel testing on a prior configuration or project, they may already be stored on a magnetic tape.

6.4 Formation of an Aerodynamic Description

For an aircraft which is not rigid, the aerodynamic description should, ideally, be uniquely related to the distortions and motions of the aircraft, defined at some set of discrete and aerodynamically relevant points. However the deficiencies of aerodynamic data, currently make it necessary to produce, more or less, independent descriptions for the rigid and elastic aeroplane.

The preceding review of data sources indicates that aerodynamic data are available at three different levels of detail, namely, pressure information, component forces and moments and overall aircraft forces and moments. To obtain a description of the rigid aircraft in the desired form, requires aerodynamic data to be available at the level of detail provided by pressure information, but such data are not normally considered to be the most reliable. If overall aircraft measurements are available these set the standard, if not, it will be set by component values defined empirically (or sometimes theoretically). A consistent description of the aircraft is attained by ensuring compatibility between the forces and moments of the overall aircraft and its components (adjusted if necessary), together with component forces and moments provided by integration of suitably modified component pressure information. The key to this process is the ability to modify the pressure distributions in a plausible fashion. If accurate measured pressure information exists then compatibility will usually be achieved by relatively minor adjustments. If only theoretical or empirical pressure information is available, this process becomes a completely arbitrary and often difficult task. This is particularly so for high incidences, transonic conditions or for any other non-potential flow condition.

For the elastic aeroplane, it follows from the review of data sources, that the aerodynamic description must usually be entirely theoretical, but modifications to the basic theoretical data may sometimes be made, to give some degree of compatibility with measured or empirical data, for the rigid components. When applied to unsteady theory these component data may be either steady or unsteady in origin. It is not usually possible to obtain complete compatibility, without attempting modification of the theoretical pressure distributions and this is not normally done unless it is absolutely necessary, or measured pressure data are available. An alternative approach towards compatibility is sometimes adopted, using strip-type methods and modified two dimensional derivatives. This approach may also be used, if no theory exists for a particular component, or as a means of studying problem sensitivity to aerodynamic variations in flutter work. If strip-type methods are used, however, the desired form of aerodynamic description is not automatically available. It should be noted, that because the aerodynamic description of the elastic aeroplane is usually of theoretical origin, it is subject to all the associated theoretical limitations.

When the aerodynamic characteristics of the elastic aeroplane are not linear functions of the distortion, it is no longer possible to consider the descriptions of the rigid and elastic aeroplane separately. It then becomes necessary to define non-linear aerodynamic characteristics which are valid for rigid aircraft motions and motions and distortions of the elastic aircraft: currently this is virtually impossible. Other notable aerodynamic deficiencies are the dearth of quantitative data suitable for the prediction of buffeting phenomena and the limited nature of data available on transient aerodynamic conditions. Yet another item, which has not been reviewed in this section, is aerodynamic data associated with drag. Such data are not available in detail form, so that only restricted loading information involving drag forces, can be estimated.

It must also be emphasised, that even when good wind-tunnel measurements are available for the correct configuration, this does not guarantee a true description of the full-scale aircraft. Apart from scale effects, there are additional wind-tunnel effects, such as transition fix, blockage, flow variations and wall effects, which it may not be possible to eliminate or correct for. Unfortunately all these effects are most likely to produce data errors in measurements at the transonic and/or high incidence conditions which are relevant to critical quasi-static loading cases.

7. THE RELATIONSHIP BETWEEN DATA STANDARDS, APPLICATION AND REQUIREMENTS

7.1 Technical Aspects

7.1.1 Aerodynamic Data

The deficiencies of aerodynamic data, in some respects, are very apparent, but when using them for certain types of investigation several mitigating factors can be invoked. The least exacting requirement is associated with flutter where theoretical methods are used almost exclusively, despite their limitations. There is, however, some justification for this, because the more significant problems tend to be dominated by the aerodynamic contributions from the surfaces, with other contributions being of minor importance. Flutter also conforms with the theoretical

limitations of small perturbations and oscillatory motion (at the critical flutter speed). Details of pressure distribution are not too important, since only generalised aerodynamic forces are required i.e. the forces associated with modes of deformation of a complete component. Other uncertainties such as high incidence effects and, to some extent, even transonic characteristics, are implicitly catered for by speed margins. This does not absolve one from assessing the sensitivity of these margins to possible aerodynamic variations, but such exercises are basically qualitative in nature.

For assessment of the stability and control aspects of the quasi-rigid aeroplane, the aerodynamic requirements are more stringent than for flutter, but they are not too difficult to meet. The aerodynamic description of the overall rigid aircraft is usually obtained from measured or well established empirical data. Aeroelastic effects are perforce based on theory and can therefore be applied as corrections to the rigid aircraft description. As for flutter, this is reasonably well justified, because these effects will be dominated by surface contributions and detail discrepancies in pressure distribution usually have only minor effects on the resultant overall description of the quasi-rigid aeroplane. This latter statement is not always quite so well justified for spoilers or trailing-edge control surfaces. Stability and control investigations are also similar to flutter investigations in being, basically, of a qualitative nature. It is only necessary to establish that control powers and handling qualities are adequate and that the aircraft is stable with adequate stability margins.

Response investigations of the quasi-rigid aeroplane for loading purposes, place far more stringent requirements on the aerodynamic description. Pressure distribution details become important and assumptions of linearity for the aeroelastic aerodynamic contributions, are far more suspect, at the high incidence conditions associated with design cases. There is also a fundamental difference between loading and stability and control work since quantitative, realistic information is demanded from the former whilst, from the latter, only a qualitative assessment is required.

The determination of accurate loading data from the response of the elastic aeroplane, can produce the most exacting aerodynamic description requirement. Compared to the quasi-rigid representation one is relying solely on theoretically determined pressure distributions. The restrictions of small oscillatory perturbations will, nevertheless, still be satisfied for small oscillatory inputs. Providing the aircraft incidences are not too high, theory can therefore still provide a reasonably valid representation for some fatigue loading conditions. Discrepancies in the estimated loads will, in any case, not be as significant as the uncertainties inherent in the methods of fatigue life estimation. For some types of oscillatory aerodynamic loading, such as buffet, because it is difficult to obtain any satisfactory input information, little meaningful work can be done. For transient types of input, particularly if they are large, the aerodynamic description of the elastic aircraft is no longer very adequate. If such problems provide a design case for any part of the aircraft, then the predicted loads need to be used with some caution. In general, however, the aircraft is almost entirely designed on loads produced by the response of the quasi-rigid aeroplane, so that the most exacting aerodynamic data requirement is identified with the prediction of quasi-static loading.

7.1.2 Mass Elastic and Systems Data - Without A.F.C.S.

Without automatic flying control systems (A.F.C.S.) the least exacting requirement for mass, elastic and systems data is associated with aeroelastics for stability and control purposes. The assumption of linear elastic properties for the structural components is reasonable and the likely order of stiffness reductions, due to temperature effects and high and low loading, will produce little change to the aircraft's stability or response characteristics. Stiffness and mass details are generally unimportant, with the possible exception of attachment stiffnesses, particularly those involving the flying control systems. However, even for these items, it is only necessary to qualitatively demonstrate that margins are adequate for the possible range of linearised elastic values. The determination of loads from the response of the quasi-rigid aeroplane, places greater demands on the accuracy of detail mass properties and, indirectly, may do so on stiffness properties, because of the effects of local distortions on the aerodynamic pressure distributions. If panel buckling occurs, this could be important, as could flying control system nonlinearities. The system response characteristics due to control surface inputs will, however, often be overruled by mandatory design requirements e.g. 'assume maximum rate of application'.

In flutter work, a situation parallel to that for the aerodynamic data exists for the mass and elastic data, since only generalised masses and stiffnesses for component mode shapes are required. Detail mass and stiffness data are, therefore, generally not so important for the structural components. Attachment stiffnesses, including flying control systems effects, can often be very important, but the branch mode approach to flutter problems is orientated towards the investigation of stiffness effects to ensure adequate margins. Thus, assumptions of linearity for the elastic properties are acceptable and certainly far less sweeping than for the aerodynamic characteristics. Large concentrated masses attached to the

aerodynamic surfaces can also be significant and, if these are present, the details of the mode shapes, at the point of attachment need to be reasonably accurate. The deficiencies of structural damping data are not too important, since flutter clearances will normally be established assuming none is present. On the comparatively rare occasions when it is invoked, the minimum likely value is used.

For dynamic loading work, as for quasi-static loading work, details of mass and stiffness can be important. Flying control system non-linearities may also need to be taken into account but the greatest potential shortcoming is the lack of quantitative data on structural damping characteristics. If there are any dynamic loading problems, they may well be associated with a poorly damped mode in which structural damping dominates. Nevertheless, the more exacting overall data requirements are still associated with quasi-static loading (for the reasons given in the previous sub-section on aerodynamic data) even though the requisite standards are easier to achieve.

The influence of fuel sloshing has not been mentioned in the above discussions of mass data, although they could obviously be relevant to every type of investigation. To include these effects properly is, currently, hardly possible, but it cannot be categorically stated that they can be ignored, even if they are very unlikely to cause problems on the typical configuration. The lack of problems results from the simple, though arbitrary, approach of introducing fuel tank compartmentation to control the static fuel centre of gravity position and inertia, and hence achieve stability of the quasi-rigid and elastic aeroplane (with mass properties defined by these static quantities). For external jettisonable fuel tanks the above static properties may be similarly controlled to obtain satisfactory jettison characteristics. This may be neither very precise, nor the optimum design solution, but it probably ensures a satisfactory aircraft whilst leaving the difficulties of defining a correct fuel mass representation unresolved.

7.1.3 Mass, Elastic and Systems Data - With A.F.C.S.

The inclusion of A.F.C.S. obviously requires a far more detailed systems description, in the form of additional equations (often non-linear) governing the motion of the aircraft. Non-linear effects e.g. authority limits need to be retained for both stability and control and quasi-static loading purposes, but it should be possible to omit high order frequency effects from the system's description. When the system is operative, a theoretical definition may be adequate for the quasi-rigid aeroplane, because it should conform to design requirements established for this representation. For loading cases associated with a closed-loop system malfunction, the theoretical description may no longer be adequate, but for open-loop malfunctions, no A.F.C.S. description is required and the magnitude of the control surface inputs is simply dictated by the authority limits. Even with the system operative, it is also possible, that certain design loads may be unaffected by the presence of the A.F.C.S., either because they occur in a particular steady state condition or because the authority limits are overridden.

The elastic aeroplane requires a system description applicable at high frequencies (the off-design state) and it is consequently more difficult to produce an adequate definition. The response of the A.F.C.S. is governed by acceleration and velocity inputs to the sensors, so that details of mode shape at the sensor location are important. Moreover, most sensors are located within the fuselage, which is the most difficult structural component for which to define accurate mode shapes; this difficulty is increased for rate gyros which require angular information. Stability assessment of the elastic aeroplane with A.F.C.S. necessitates linearisation of the system description and evaluation of gain and phase margins throughout the loop. If linearisation is not really adequate, response investigations are necessary for a large order problem governed by a non-linear set of equations. Providing that only a few of the lower frequency modes of the basic aircraft are effected by the A.F.C.S. and no marked changes are produced, stability assessment with a linearised theoretical description may be adequate. This description should then be adequate for small amplitude periodic dynamic loading problems. Large amplitude transient problems can be accommodated, if the authority limits are overridden. If the above provisos are not satisfied, a non-linear and probably non-theoretical system description will be necessary. A description of this latter form is likely to be required for the study of closed-loop system malfunctions. As for the quasi-rigid aeroplane, open-loop malfunctions do not require an A.F.C.S. description and any associated periodic dynamic loading information is governed by control surface inputs between the authority limits.

Because of the compounded uncertainties of the data from the various sources required for stability and response investigations on the elastic aeroplane with A.F.C.S., this is considered to produce the most exacting requirement for systems data. Instabilities and dynamic loading cases may, or may not, exist, but it can be a very considerable task to even assess the situation. It is perhaps better practice to design a good basic aircraft and thereby avoid the need for high-gain systems, to minimise the chances of such problems occurring.

7.2 The Initial Design Stages

The start of a project is typified by a series of short duration configurations leading to one of long duration. This is then followed by interspersed short and long duration configurations, culminating in a firm design. For the early short duration configurations, an aerodynamic shape is drawn to accommodate the necessary internal installations, possible forms of structure examined and overall weight and centre of gravity estimates made. Performance (including high-lift) and 'stability and control' design aspects will largely control the aerodynamic shape at this stage, but some rudimentary quasi-static loading information may be requested and aeroelastic advice sought. These loads will be based on theoretical and/or empirical data and specified at the level of detail set for the short duration configuration. Assessments of the resultant configuration will also be made for all other relevant design aspects and the whole process repeated for several configurations, until the first long duration configuration emerges. A major effort, on the tasks of load estimation and aeroelasticity, can then proceed. This event will be marked by the production of one, or more, rigid wind-tunnel models aimed at measuring steady quantities for performance and 'stability and control' design aspects. At the same time, some corresponding structural and flying control systems design will be in the process of definition. Theoretical or empirical analyses of the various items, as they are designed, will then progressively provide elastic data, together with detail mass data. The quality of these data will be largely set by the quality of the prior loading information on which the design was based. Only elastic systems data will usually exist at this time, as the main emphasis, during the initial design stages, is on the definition of the necessary system parameters. The relevant aerodynamic data will be obtained from the test results of the wind-tunnel models (when these results become available), and theoretical and empirical sources.

If testing is confined to models of the form specified, then for load estimation and aeroelastic purposes where existing theories are inadequate, the standard of these aerodynamic data is obviously inferior to the corresponding elastic and mass data, as well as being more restrictive in its application. An improvement to the standard can be achieved, if an early decision is made to build further wind-tunnel models, specifically for the purposes of load estimation and aeroelasticity. Thus component load models and pressure models may sometimes be built to the lines of the first, or a subsequent, long duration configuration: these models will be rigid and will measure steady quantities. Even though such additional models may be built, it is unlikely that the project configuration will still be same, when the test results become available. It is therefore almost inevitable, that all the wind-tunnel results, obtained during this period of project design, must be classified as empirical data. Furthermore, unless one is prepared to assume that certain aerodynamic quantities are insensitive to the variations of configuration being studied, then the test results need to be expressable in a parametric form for further use. The parametric form should also, preferably, permit correlation with data from other empirical sources. Thus whilst a better estimate of data, essentially for quasi-static loading purposes, is obtained with the additional models, the data still have many deficiencies.

Flutter models are sometimes built during the initial design stages, as well. These again are likely to be outdated by the time that test results become available so that empirical correlation or theoretical matching techniques are necessary to make use of the information for a different configuration. Flutter models can, however, fulfil another qualitative function, by indicating which flutter problems are likely to be the more significant. As the design proceeds further models (or modifications to existing models), for performance and stability and control purposes, will often be tested but no parallel developments will normally occur for loading models. Some guidance on empirical procedures for modifying aerodynamic data for load estimation is, nevertheless, provided by the test results of these other later models.

During the initial design stages, the rapid provision of adequate information is essential, if an effective contribution is to be made towards the design. Now, many of the requisite theories for acquiring input data, and the subsequent processes for producing stability and response solutions, are totally dependent on the digital computer (particularly for aircraft of the advanced combat type). Fulfilment of the dual requirements of speed and adequacy is thus automatically provided by use of the computer. Even when this total dependence on the computer does not exist, it is still both advantageous and economic (unless very large manpower resources exist) to utilise the computer for calculations of a repetitive or routine nature. Since it is virtually always possible to obtain digital solutions, given the necessary input data, any failures to meet the requirements of speed and adequacy will be confined to methods of data acquisition.

Of the required data, failures therefore tend to be associated with non-theoretical methods of acquisition, which do not utilise the digital computer. Once again, it is the data needed for quasi-static load estimation that proves unsatisfactory, because of the time taken for data acquisition and processing. If wind-tunnel test data are not available, then the data must come from other sources and will require fairly extensive and time consuming manual manipulation to produce an aircraft description in digital form, suitable for subsequent digital operations. When test data are available, the standard of data is increased but so too is the quantity, particularly for the loading type of model. So that, apart from the inevitable delay caused by the time needed to build and test a model, a further delay is introduced, because of the time needed to digest and process all the data produced. Current digital processing on such data, mainly covers data presentation and does not extend into

the subsequent processes needed to define an aerodynamic description of the requisite form.

7.3 The Specification and Design Requirements

The specification and design requirements influence all technical aspects of aircraft design but load estimation is one of the areas for which fairly rigorous quantitative requirements are formulated. There are obvious advantages in this, in that it removes the onus and responsibility of specifying suitable design criteria from the industry. However, quantitative requirements demand quantitative information and, if one presumes that the requirements are realistic, the information supplied must also be realistic. This emphasis on realism for loading information is necessary if an aircraft is to be safe and at the same time not be compromised by excessive weight penalties. If this philosophy is accepted, then the necessary standards of data accuracy can obviously be affected by the specification and design requirements. Aerodynamic data are probably the data most affected in this way, because of the uncertainties often associated with many of the critical loading conditions, when even wind-tunnel measurements must be regarded with some suspicion.

When asked to produce realistic loading information in these circumstances, one would like, therefore, to feel confident that the demands resulted from genuinely realistic requirements. Thus, for example, when the design flight envelope is specified for an aircraft, a firm distinction should be drawn between strictly necessary and desirable characteristics; the corner points of that envelope should be related to practicable overshoot conditions rather than dictated by an arbitrary formula or the desire to produce an aesthetically pleasing diagram. In addition, assurance would be welcome that the origin of some of the loading requirements concerned with pilot inputs to the control surfaces are not related to a different generation of aircraft and possibly pilots as well. In the early design stages, it is also impracticable to ask for loading information which is very dependent on details of the aircraft description: the standard of the information produced is no better than the standard of data used to make the necessary calculations. The above situation can occur in certain problems of dynamic load estimation.

8. CONCLUSIONS AND RECOMMENDATIONS

The main conclusion of this report is, that the deficiencies of aerodynamic data, particularly for quasi-static loading purposes, produce the greatest obstacles in the activities of load estimation and aeroelasticity. The inadequacies of such data are highlighted by the operational requirements and resultant configuration, typical of an advanced combat aircraft, and by the initial design stages of a project. Fundamentally, the inadequacies stem from the very restrictive limitations of aerodynamic theory: some of these limitations reflect the current state of the art but others are the outcome of restrictive approaches. Any aerodynamic theory developed should be usable for loading and/or aeroelastic work, enabling, ultimately, the determination of loads and moments on a general quadrilateral panel, together with consistent pressures at any points on that panel. Aerodynamic theories for aeroelastic applications should be formulated in terms of distortions which are more typical of a real structure, rather than overall polynomial nodes (for example) or undefined distortions, lost in the methods of numerical analysis. Theoretical aerodynamicists should also be far more concerned with the real, although perhaps difficult, configurational features of an actual aeroplane and the development of methods of aerodynamic idealisation should receive the same sort of consideration, as that given to structural idealisation. In addition, most of the real aerodynamic problems requiring attention are associated, perhaps unfortunately for the theoretician, with transonic and/or high incidence conditions.

Since it is very unlikely that the necessary improvements in aerodynamic theory will occur rapidly, means of improving the standard of data and the processes of acquisition from measured and empirical sources, also need to be considered. The conventional rigid wind-tunnel model measuring overall forces and moments has diminished utility for an advanced combat aircraft since, in many of the critical regions of the flight envelope, quasi-static aeroelastic losses, ranging from 10% to 75% on different parts of the aircraft, will usually be present.

The possible alternative of building models, capable of producing representative quasi-static aeroelastic distortions, is not a practicable one, if adequate coverage is to be given to the necessary range of flight configurations and conditions. Models can, however, play a part in the development of suitable quasi-static aeroelastic methods, which would, ideally, be basically theoretical with empirical modifications when necessary. Since the test results from a rigid model, measuring overall forces and moments, must be empirically broken down into components, prior to the determination and incorporation of quasi-static aeroelastic effects, it would seem a logical step forward to use rigid component load models from the outset, even if some limitation on the number of component measurements has to be accepted. The information provided by more general use of such component load models, would also provide an additional fund of empirical data for loading purposes. This would assist the development of empirical data sources and methods, to provide a consistent counterpart to 'stability and control' data sheet methods. It is also concluded that, when theoretical methods are unsuitable, the digital computer needs to be used to a greater extent, for the storage and subsequent processing of data to produce aerodynamic descriptions with the necessary speed and consistency. When suitable theoretical methods later become available, if large computer times are required to produce information for a single condition, it may still be economic to treat these data in a similar fashion to that devised for empirical and measured data.

Other notable data deficiencies exist in the representations of fuel mass and automatic flying control systems, particularly for the frequency range associated with the elastic aeroplane. However, the correct representation of fuel mass would be unlikely to provide any critical problems on an aircraft which adheres to the conventional design procedures, whilst the standard of systems data is not too critical in the initial design stages. The subsequent solution of the stability and response equations, which are necessary to the tasks of load estimation and aeroelasticity, can be rapidly accomplished to an adequate standard by using digital computation.

The design of an advanced combat aircraft, strongly emphasises the integrated nature of work which involves several different technical disciplines. The boundaries between stability and control aspects, quasi-static loading, flying control systems analysis, flutter, dynamic loading and stressing cannot be clearly defined, especially when the aerodynamic characteristics are not linear functions of aircraft motions and/or distortions. Particular examples of interdependence, of direct interest to advanced combat aircraft design, concern the formulation and solution of the equations of motion for the quasi-rigid aeroplane with non-linear aerodynamic characteristics (as above) and the definition of suitable stressing parameters for monitoring loads, during transient responses of both the quasi-rigid and elastic representations of the aeroplane. A policy of segregating the work into isolated specialist sections is therefore not conducive to the rapid and consistent appraisal of a proposed design. Groups of people, capable of dealing with several disciplines are needed, with a level of specialist knowledge not much below that already existing within specialist areas. This is not an insuperable objective, since, as this report has indicated, many of the tasks with aerodynamic associations are only basically concerned with assessments of aircraft stability and/or response characteristics. One of the main handicaps to progress in this direction is specialist jargon, which in some cases is a product of outmoded conceptions and methods.

There is also an important relationship between the operational and design requirements, which control the specification of loading information, and the standards of data which can be achieved. The acquisition of adequate aerodynamic data, for the prediction of realistic loads, can often prove difficult and the resultant effort serves little purpose if unrealistic, or unnecessarily severe, requirements are imposed. If realism is not the objective, then much of this report becomes irrelevant, since improvements in technique or even existing techniques are unjustifiable elaboration. A further type of unrealistic requirement in the initial design stages (when data standards are not completely adequate) concerns requests for dynamic loading information, likely to be sensitive to the elastic representation of the aeroplane.

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THE INFLUENCE OF NEW MATERIALS ON AIRCRAFT DESIGN

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SUMMARY

An investigation of military aircraft characteristics and performance trends indicates that important advances have been made in almost all aspects and that orderly evolutionary changes should continue. Currently available materials are reviewed and their limitations and future potentials discussed. New materials and their application and development programs, which were initiated to validated their design potentials as well as to establish confidence in their structural impact on future military aircraft, are highlighted.

The findings are that significant potential savings will be possible, if new metals are utilized to the fullest extent, but that revolutionary advances in aircraft design and performance could be made by the utilization of the new advanced filament composite materials.

It is also noted that many potential advances have not been made because of low cost effectiveness potential, lack of confidence in the new material, extensive data requirements, the requirement for large capital investment in new machine tools and the presence of a highly skilled labor force.

Finally, the requirements for a better mutual understanding between the technical specialists, which are necessary if these new materials are to be integrated into new military aircraft, are outlined.

Sommaire

Les résultats d'une étude des caractéristiques et d'une analyse des perspectives pour le rendement futur des avions militaires indiquent que des progrès significatifs ont été fait en presque tous les domaines et qu'on doit compter avec des améliorations systématiques. Les matériaux disponibles sont catalogués et leurs limitations et leurs possibilités sont presentées. On a mis en relief des nouveaux matériaux aussi que les programmes d'exploitation et de développement correspondants. Ces programmes ont été lancés afin de mettre à l'épreuve la viabilité de leur construction et de fonder une connaissance plus detaillée sur la resonance qu'ils auront d'avoir au dessin des appareils militaires.

On a constaté qu'il sera possible de réaliser de grandes économies en utilisant toute la gamme des possibilités des métaux récemment apparus. D'ailleurs des avancements boulversants peuvent être atteints dans le domaine de la construction et de la performance des avions par l'emploi des nouveaux matériaux au filament composé avancé.

On a souligné en outre que des nombreux avancements déjà réalisables en vue de la technologie existante ont été empechés par un facteur de rendement économique réduit, la manque de confiance en ce qui concerne les nouveaux matériaux et la nécessité préalable de ramasser une quantité considérable de données, de faire de gros placements en nouveaux machines outils et de maintenir un corps de personnel très spécialisé.

Enfin on a exposé à grands traits les conditions sous lesquelles des spécialistes techniques peuvent mieux s'entendre les uns les autres, ce qu'il s'impose si les nouveaux matériaux ont d'être incorporés aux avions militaires futurs.

THE INFLUENCE OF NEW MATERIALS ON AIRCRAFT DESIGN

R. F. Hoener

1. INTRODUCTION

It has often been stated that Materials are the Key to the future. An examination of new materials and their influence on future military aircraft will, therefore, provide an insight into new military aircraft systems and indicate the key new potentials as well as highlight areas for increased emphasis.

The aircraft will be the means for this review and assessment, and the natural and induced environments to which it is exposed during its operational life time will be considered as the obstacles to be overcome.

Basically, no aircraft is any better than its structure, and no structure is any better than the materials utilized; for the materials/structures system provides the physical form of the aircraft as well as satisfying the following requirements:

Aerodynamic	
Smoothness	Erosion Resistant
Correct Contour	Distortion Resistant
Structural	
Loads Transmission	Stiffness
Maintainability	Reliability
Vehicle	
Performance	Containment
Life	Shelter

Prior to delineating the new materials and their influences, it is deemed appropriate to establish a basis for comparison by reviewing the past in order to establish prior accomplishments, capabilities and trends in military aircraft.

Since a normal aircraft development program requires four years, and since a new material requires from ten to fifteen years from the time of its conception until its wide application in aircraft, this discussion will be directed toward aircraft of the post 1975 period.

2. CAPABILITIES AND TRENDS

A review of the current and projected operational environments, development trends and potential future capabilities indicates an expanding frontier complex in scope which will pose a continuing challenge to materials/design research in the future.

The flight regimes shown by the altitude-velocity plot of Figure 1 (1) include areas of sustained flight, those considered feasible for future sustained airbreathing flight, and the broad corridor potentially available for boost glide or lifting reentry flight. Almost all flight experience to date has been confined to the subsonic regime - small solid portion; however, the area up to 50,000 feet and 2,000 feet per second has also been utilized for extensive operational military flying. Experience in the expanded envelope shown for current sustained flight has been gained with only a few aircraft, such as the B-57D, U-2, XB-70, and SR-71. Potential expansion of this regime by ramjets, scramjets, or mixed cycle engines is shown up to speeds of 12,000 feet per second. To date, only a few ramjet missiles and the X-15 have probed this regime. Expansion to even higher speeds has been studied but is currently speculative. The glide and lifting reentry regime has been probed by the X-15, ASSET, and PRIME flights. While total flight time above Mach 3 is limited to relatively few hours, significant progress has been made in developing the needed technologies through those flight programs, the X-20 (DynaSoar) development, NASA research efforts and contractor programs. Numerous studies of boost glide systems, reusable launch vehicles, and lifting reentry spacecraft have examined potential system capabilities, the structural development, and material research requiring emphasis. Although many orbital launches and several eminently successful manned reentry flights have been made, much remains to be accomplished in these areas.

On this Flight Regimes Figure - Figure 2 - there are superimposed the dynamic pressures and the equilibrium temperatures for the lower surface skins five feet aft of the leading edge of the lifting surface in order to indicate the magnitude of the induced environments encountered by the flight vehicles in the various portions of the flight regimes.

Progress as indicated by these Flight Regimes figures and the trends depicted by Figures 3 through 7 forecast the expanding frontier, scope, and complexity with which the materials specialist must cope.

Figure 3 shows the trends in design speed over the past twenty years while Figure 4 shows the trends in design gross weight over the same time span. A simple projection of the maximum gross weight trend indicates that, if the trend continues, we shall have the million pound airplane by 1980.

A relatively new environmental phenomenon is that of sonic fatigue which is induced by the high proportion of the energies in the exhausts of the propulsion systems or in high speed flows that are converted to noise of sufficient intensity to fail the structure and to degrade the performance of equipment and the crew. The swift increase of this noise environment with time is presented in Figure 5. Simple physical considerations indicate that near field noise for limited areas of VTOL vehicles can reach 170 decibels caused by the doubling of noise sound pressure levels by full ground reflection.

Shown in Figure 6 (2) is the trend in airframe materials application which is influenced by the loading intensities, the environmental effects, the improvement of the materials' characteristics, manufacturing capabilities and costs.

The universal yardstick for measuring structural efficiency has been the structural weight fraction (λ_s), or the ratio of the structural weight to the take-off gross weight. In the 1938 edition of their Airplane Structures, Niles and Newell noted that ...... "structural weight of landplanes varies from 25 to 35 percent of the total". The trend, slightly downward in this measure of efficiency, as shown in Figure 7, is a tribute to the designers who in the face of the requirement for the increased complexity and the more severe operational environment (Figures 2 through 6) have been able to make such an improvement possible partially through the use of improved materials but primarily through new design concepts. The only airframe that apparently does not comply with the trend is one which not only incorporates most of the features which tend to increase the structural weight but also is of a size never before attempted in airplane design and is the only one - since it has only recently flown - which has not benefited from potential growth.

The reduction of the structural weight fraction is one of the most significant means of increasing aircraft operational efficiency. Since payloads vary from 2 to 30% of the gross weight, depending on fuel load, flight distance requirements, and vehicle type; the reduction of the structural fraction by 40% could result in sizable increases in the payload fraction, 25% in one case and 400% in another; for example.

And structural weight savings of this magnitude will be attainable by the utilization of the new materials and design concepts to be discussed later.

Prior to investigating the influence of new materials on aircraft design a review of existing materials and materials development programs will be made in order to establish a basis for comparison. An example of the materials employed in a recent aircraft design by type and fabrication process is shown in Figure 8 (3).

3. MATERIALS REVIEW

3.1 Aluminum

Aluminum alloys are the backbone of many aerospace vehicle structures and provide the designer with a broad spectrum of materials properties. In general, aluminum alloys suffer from three primary deficiencies, namely low fracture toughness, stress corrosion susceptibility and poor weldability, in various degrees, at high strength depending upon alloy composition and heat treatment.

These problems are being attacked and advances have occurred in the development of improved stress corrosion resistant alloys by the use of new thermal treatments such as overaging to the T73 condition. The 7075 and 7079 plate and forging alloys in the overaged T73 condition have largely replaced the higher strength tempers because the improved stress corrosion resistance is more valuable than the 12 to 15 percent loss in strength. For large parts procedures for selective overaging of the stress corrosion critical areas are being developed. Development efforts on alloy additions which will provide the requisite stress corrosion resistance and the higher strength are being vigorously pursued.

A large inventory of alloys exists with wide latitude for trade-offs in material properties. It is anticipated, however, that only modest increases in strength will be attained, but significant improvements will be made in welded joint strengths, ductility, fatigue and stress corrosion resistance.

3.2 Titanium

Titanium has been established as a prime flight vehicle structural material having been used in the F-100 (Aft fuselage), the XB-70 (fuselage), the Mercury (Heat shingles) and the SR-71 as well as in commercial aircraft. Titanium alloys in the alpha, alpha-beta and beta phases exhibit a wide range of desirable physical properties with the beta alloys having the highest yield strength with high elongation and the alpha alloys having a wider temperature range of good ductility and toughness.

Requirements for sustained operation at cryogenic temperature, as well a moderately-high-temperatures, will result in the wide application of titanium to such structures, as it shows definite strength and ductility advantage in both areas. It is possible to achieve high strength/weight ratios with titanium alloys, and the lighter weight permits increased geometrical cross section for a given weight, resulting in superior performance under compressive buckling. Titanium is resistant to atmospheric and salt water corrosion and completely compatible with most common

fuels and oxidizers. In addition, titanium alloys are fabricable, many are weldable and they generally provide good fatigue properties.

A wide variety of alloys are available for use over the temperature range from -423 to 950°F. Research is aimed at developing a stable alloy for extended use up to 1200°F and in improving the strength, ductility, fracture toughness and stress corrosion of current materials.

In 1968, however, Titanium Metals Corp. of America reported that total mill product shipments were only 12,000 tons - approximately 14 percent below the previous year - and substantially below the expanded industry capacity.

3.3 Steels

Steels have found and will continue to find a wide application in the aerospace vehicles because of their ready availability in many forms and shapes, ease of fabrication, wide range of physical properties, low costs and broad base of applied knowledge concerning their application, wide range in temperature capabilities, and the intensive research performed by almost all industries to improve their properties.

Maraging steels are a new family of alloys which possess a combination of high strength and fracture toughness that cannot be achieved with current high strength, low-alloy steels and should find wide application. Two deficiencies in these alloys are low fracture toughness in weldments and delamination in thick sections which must be overcome.

Air Force Material Laboratory (AFML) research has produced a martensitic high strength, high temperature, corrosion resistant steel known as AFC-77. This steel,which is the strongest of the current metals, can be used to replace more costly superalloys below 1200°F as well as to replace ferritic stainless steels when higher strength and improved oxidation resistance are required.

Alloys with a wide range of properties are available. Current effort is directed at increasing the reliability of available steels by improving their toughness and resistance to environmentally-induced delayed-failure mechanisms such as stress corrosion cracking, hydrogen stress cracking and surface adsorption effects.

3.4 Beryllium

Beryllium is finding its place, although a limited one, in the structures of launch and space vehicles. The properties which make this material so attractive to flight vehicle structures are its high modulus of elasticity, low density, high melting point, good thermal conductivity, and high thermal capacity. The biggest single deterent to the wide spread application of this material in the aerospace industry has been its inherent brittleness. Alloying with aluminum has produced improved ductility but at a sacrifice of other properties.

This inherent deficiency has now been accepted and has resulted in increased studies of "design with brittle materials". Current advanced development programs are directed toward determining the extent of possible application of beryllium in both primary structure and propulsion systems. These results will influence the degree of future use.

Beryllium is commercially available, and future effort will be concentrated on increasing its reliability and reproducibility by control of purity and production processes.

3.5 Superalloys

Alloys of nickel and cobalt are available for high temperature applications and have been utilized extensively in gas turbine engines and high performance aircraft structures because of their good strength and oxidation resistance in the temperature range 1200-1800°F.

The discovery that the addition of thoria particles to nickel and cobalt produced increased stability at high temperature promises significant advancement in superalloys. The use of this dispersion strengthening mechanism together with alloying elements such as chrominum and molybdenum have resulted in experimental alloys with improved strength and oxidation resistance at temperatures up to 2200°F at low to moderate stress levels. Development of oxidation resistant coatings is in progress and will be essential for long-time component integrity.

Conventional alloys are available for use up to 1800°F for structural and propulsion applications. Continuing research is expected to produce further increases in strength and oxidation resistance by dispersion strengthening and alloying to permit higher temperature operational levels.

A new superalloy, Rene' 95, has been developed for the AFML. This new alloy has exceptionally high strength in the intermediate temperature range - tensile yield 190,000 psi at 1000°F - which will provide an estimated 25 percent savings in weight in turbine engine compressor discs.

3.6 Refractory Metals

Columbium, molybdenum, tantalum and tungsten, because of their high temperature melting points and their strength capabilities above the temperature limits of the superalloys, have been exploited for use in radiative heat shields, rocket nozzles and structural elements in hypersonic cruise and

reentry vehicles. Their primary limitation is poor oxidation resistance at high temperature requiring a protective coating. Many of the early difficulties in processing, forming, fabricating and joining these materials have been overcome. The feasibility of using and reusing columbium and molybdenum in reentry structure has been proven by the ASSET flights. Tungsten has shown its merit as a rocket nozzle insert and in high temperature fasteners.

Alloys, with protective coatings, are available for use up to 3000°F. Continued investigation of alloying additions and strengthening mechanisms will provide improvements in mechanical properties at temperatures, with emphasis on columbium due to its versatility for a wide variety of applications; and tantalum because of its ease of fabrication and weldability.

3.7 Coatings

Coatings to protect refractory metals from oxidation at high temperature are mandatory. Coating systems have been developed which afford protection from one hour to one hundred hours dependent upon the alloy, temperature and environment. Coating development for the refractory metals can best be described as at the advanced laboratory state, ranging from semi-commercial coatings for molybdenum; through pilot-work on columbium; to promising developments for tantalum and tungsten.

The development of fused slurry silicide coating systems has provided a quantum jump in coated refractory metal alloy life. Fifty or more reuses of reentry vehicle heat shield applications are now possible.

3.8 Ablatives

Ablative materials, which have alleviated the thermal problem in hypersonic flight, have a number of inherent advantages such as; light weight, low cost, ease of fabrication, passive in operation, resistant to high heat fluxes, insulate substructure, and no upper limit to service temperature. Among their limitations, which are few, are decreased efficiency with long exposure time, out gassing in vacuum, and susceptibility to mechanical damage.

The large number of materials which have been developed for ablative uses have been utilized in a homogeneous state or as a composite material. Homogeneous materials include: epoxy, phenolic, polyamide, polyurethane, polytetraflouroethylene and pyrolyzed resins. The composite materials are further classified as reinforced plastics - organic or inorganic resins containing various reinforcing agents, filled plastics, organic or inorganic resins containing powdered fillers, and internally ablative plastics - low temperature ablating resin in a charred resin matrix, charring resins impregnated with a subliming endothermic salt, low-temperature ablating resin in a porous ceramic matrix.

Although the current ablative materials have performed well efforts are underway to: provide low modulus polymers by adding plasticizers to the basic polymer, extending the polymer chain, or copolymerizing the resin with other elastomerics polymers; improve the char strength of phenolic resins by increasing the carbon content of the phenolics by copolymerization; increase the molecular weight of moldable polyphenylenes, improve their thermosetting characteristics, provide new curing agents; and develop organoboron resins as ablative materials.

3.9 Filament Composites

The previously discussed materials have made and will continue to make possible significant evolutionary advances in aircraft design. The filament composites which are the subject of this section will provide for revolutionary advances in aircraft design because of their high strength, great stiffness and low densities.

Glass fiber reinforced plastics have long been used for secondary structures, radomes, antennas, pressure vessels, fluid containers and rocket motor cases. They are characterized as having excellent fatigue life, high strength to weight ratios and as being non-corrosive. In addition, they can be "tailor-made" to specifically meet various loading and environmental conditions. Initial efforts with these materials had been based on glass filaments or cloth as the reinforcing agent in phenolic, silicone or epoxy resin matrices.

Recent advances in both reinforcements such as boron and carbon filaments and in matrix materials such as polyimide and polybenzimidizole resins will create a completely new set of materials and with a vastly increased scope of applications. Reinforcement of ceramic and metal matrices is also being accomplished in order to produce higher strength to weight, temperature resistant and shock resistant composite materials.

A broad and expanding base of composite technology for structural applications currently exists. Present studies and manufacturing technology programs are directed to analysis, design and fabrication efforts to more fully explore and extend the application and use of conventional composites. Advanced composites are being rapidly exploited by the development of engineering materials and hardware demonstration.

A summary of material usage ratings and costs is contained in Table 1 (4).

4. APPLICATION PROGRAMS

4.1 Beryllium

To acquire necessary test and operational experience and to provide confidence which will lead to applications exploiting beryllium's very favorable high strength-to-weight and modulus of elasticity-to-weight ratios, high thermal conductivity and specific heat, two programs were undertaken to provide design data and structural concepts, to improve production and processing techniques, and to perform evaluation tests.

In the first program the contractor (3) designed, produced and tested two box beams to the requirements of an aerospace plane type vehicle. The wing box structures were designed for operation in the -65°F to 500°F temperature range with representative external loading conditions.

The first of these boxes failed at 156% of limit load in a section approximately two inches outboard of the root fitting with the initial failure attributed to a local compressive instability in the upper surface skin and stringers. The test temperature was -65°F. The second of these boxes failed at 150% of limit load for maximum temperature gradients at a significant time after application of a maximum temperature gradient. The failure was initiated by a post buckling shear type failure in the outer panel front spar web when the stabilizing tensile stresses in the web began to decrease with the decrease in temperature gradient.

In the second program, the contractor (4) designed, produced and tested a rudder for the F-4C airplane. The rudder, Figure 9 (4), consists of all beryllium parts aft of the hinge line with the exception of the aluminum honeycomb core, the mechanical fasteners, and some shear clips. The core and fasteners are not available in beryllium, and the cost of the clips was considered too high for the weight saved. No attempt was made to utilize beryllium forward of the hinge line because a portion of the weight saved would only have been placed back into the balance weights. This rudder, which resulted in an overall weight saving of 40% (37.5 pounds versus 63.03 pounds for aluminum), has satisfactorily passed all test requirements for: (1) balance weight fatigue - 50,000 cycles at 40g; (2) a rolling pull-out flight condition in which both the fin and rudder supported 150% of the limit load; and (3) full available actuator hinge moment condition with the airload cp at 30% of the chord.

Two tests were conducted for condition number 3. The first test was discontinued at 150% of the limit load. Since the aluminum rudder had previously sustained 225% of the limit load, a second test of the beryllium rudder was performed in order to obtain a direct comparison. At 205% of limit load a crack occurred at the forward end of the trailing edge lower closure rib. However, the applied load did not fall off as a result of this crack. The test was, therefore, continued to 250% of limit load without further incident.

This rudder was subsequently installed in the Sonic Fatigue Facility at WPAFB and sustained 12 hours of test at 150db and 10 hours of 153db without serious consequences but experienced a sonic fatigue failure in the lower torque box skins and ribs after 45 minutes exposure to 156db. An x-ray taken prior to sonic testing indicated the presence of a crack in a rib web to flange radius which was the result of the static testing. Subsequent x-rays taken at 4, 8, 12, 16, 19 and 22 hours showed the growth of this crack and additional damage. This performance exceeded the requirements. After completion of the sonic tests, a residual strength test for the loading of condition 3 was conducted. The rudder, in the damaged condition, momentarily supported 200% of the limit load.

In order to further evaluate this material and to prove its adequacy for combat aircraft, surplus small panels were subjected to gunfire tests with satisfactory results.

Based on the successes of this program another beryllium rudder was fabricated for flight test evaluation which was started in May 1968.

While performing a rolling pull out maneuvering in the tenth hour of the flight testing, the pilot lost control as the result of the aircraft becoming roll-yaw coupled and two 360° rolls were made before the pilot gained control. The strains recorded during this inadvertent maneuver indicated that both the fin and rudder were subjected to loads in excess of design. The rudder, in fact, was permanently buckled (12 inches long, 5 inches wide and .04 inches deep) on the right side with the buckle extending diagonally down from the lower hinge. Since no other damage was incurred and since the buckle did not detract from the ability of the rudder to support the design loads, flight testing was continued. At the time of this report (1 May 69) fifty three hours and eighteen minutes of flight time have been accumulated without further incident.

Studies made in conjunction with this program have shown that an additional significant savings of 770 pounds out of 2240 pounds could be made in the basic F-4C airframe by substituting beryllium for aluminum, titanium and steel - Figure 10 (7), and that savings up to 75% could be made in the control surfaces of Mach 3 aircraft.

The weight savings predicted for the main landing gear brakes, as indicated on this figure, have been verified by the contractor who successfully flight tested the brake discs on two F-4 aircraft. Another interesting application in a brake system is that for the C-5A. In this system both the stator and rotor discs on all twenty four wheels were designed for beryllium at a weight

savings of 1500 pounds for the airplane.

4.2 Dispersion Strengthened Metals

In order to efficiently and effectively bridge the gap in materials application between the superalloys and the refractory alloys in the 1400°F to 2400°F range, exploitation of the good strength and inherent oxidation resistance of dispersion strengthened metals (DSM), both nickel and cobalt based, has been initiated. A three phased effort is underway to determine representative structural components of future vehicles which could advantageously utilize these materials, establish materials acceptance requirements and develop engineering design data. To date, the more promising structural configurations have been selected, structural design data developed, preproduction components fabricated and tested, an aircraft component designed, and efficiency studies initiated.

4.3 Refractories

The refractory metals, which have high melting points, high-temperature strength, high thermal conductivity and low thermal expansion, will be the basic materials for heat shields and primary hot structure for hypersonic cruise vehicles and high L/D reentry vehicles. Unfortunately, these materials oxidize rapidly, are expensive, are difficult to fabricate and are extremely low in specific strength.

Much has been learned, however, from the several flight vehicle development programs that have provided structural material systems design concepts, manufacturing methods and design data for hypersonic flight vehicles utilizing the super-alloys, refractory metals, and ablators. Notable among these were the: X-20 (DynaSoar), ASSET, ASCEP and the PRIME reentry vehicles. Several advanced development programs now underway will provide supplementary data. Materials and types of construction for two of these programs are shown in Figures 11 and 12 (8, 9). Notable contributions made by these programs to the structural materials areas of processing, fabrication, and testing of:

REFRACTORY ALLOYS	REFRACTORY ALLOY COATINGS
CERAMICS	REFRACTORY ALLOY FASTENERS
SUPERALLOYS	HIGH TEMPERATURE INSULATION
CYROGENIC INSULATION	BRAZING ALLOYS

Much more must be learned about the processing, joining, coating and long term characteristics of these most interesting materials prior to their utilization in manned flight vehicles.

4.4 Composites

Recognizing the need for a concerted effort in order to gain engineering design data, to develop hardware, to establish manufacturing techniques and machinery, and to instill confidence in the technical community for these materials as well as to insure its earliest possible application in aircraft design; the Air Force Materials Laboratory initiated an Advanced Development Program to accomplish those objectives for Advanced Filament Composites.

Fundamental phases of the program provide for the simultaneous investigation of the composites on the micro, macro, and engineering levels with the efforts directed by both materials and structures engineers.

The tremendous advantages of filament composites are readily apparent from the data contained in Table 2.

The initial phases of this program included the fabrication of a number of representative structural components for aircraft, helicopters, reentry vehicles and turbine engines.

For aircraft application two identical 5 foot x 5 foot load bearing components from the stabilizer of the F-111 were designed, fabricated, and tested. These components, which utilized boron-epoxy skins, full depth honeycomb core, fiber-glass closure spars along each edge, and titanium end fittings, were determined by test to be one percent stiffer under supersonic loadings and four percent stiffer under subsonic loadings than the design requirements. One component was statically tested to destruction at 133% of the limit loading (89% ultimate). The other component was successfully exposed to four life times of the fatigue spectrum and then static tested to destruction. This component failed at 113% of limit load (75% ultimate). Both failures occurred in the attachment of the composite structure to the titanium end fitting.

This program demonstrated that the computerized design techniques established for the program are adequate for hardware design, that adequate controls can be attained for the hand layup technique of fabrication, and that a projected weight payoff of 32% can be achieved.

In another airframe structural component program, two identical wing boxes representative of the T-39 center box were designed, fabricated and tested by North American Rockwell. These components - 24 inches x 16 inches x 6 inches - included upper and lower plates of boron-epoxy skin with aluminum honeycomb core, fiberglass and stainless steel test attachment fittings.

The first box was static tested to destruction with failure occurring at 90% of limit load

(60% ultimate). This failure, which occurred well below the design requirements, was attributed to the extreme difference in stiffness between the specimen and the steel test fittings which resulted in excessive shear stresses.

The second box was fatigue tested and successfully attained three life times of the service load spectrum, but failed two-thirds through the fourth life time.

The results of this program highlighted the importance of matching the design stiffness in the boron-epoxy to metal attachment areas.

Three major helicopter rotor blade components were designed, fabricated and tested by Boeing Vertol. The first, a 28 inch section of the UH-1F tail rotor constructed of boron-epoxy skins, trailing edge and spar cap with an aluminum honeycomb core and a stainless steel leading edge erosion cap, proved to be 36% stiffer in bending, 41% stiffer in torsion, and 20% lighter than its aluminum counterpart. The second, a 6 foot representative section of a main rotor blade, was made entirely of boron-epoxy and fatigue tested in the free - free beam mode. The failure which occurred after only 81,400 cycles was attributed to poor fabrication techniques which allowed a 30% void filled composite material to be produced. The third, a main rotor root end component, was fatigue tested through five million cycles of the fatigue load without failure.

In addition to these laboratory test programs, a functional flight test program is currently being performed on the F-111 airplane on a wing lower surface air deflection door 5½ feet x 1 foot with boron-epoxy skin and full depth honeycomb core, a wing panel and an aft main landing gear door. Other programs, which exploit this new material system, are currently in the planning phase. Of particular interest is the boron-epoxy F-4 rudder being developed by the contractor as a part of his Independent Research and Development program which will provide a basis for comparison with the aluminum and beryllium rudders.

A second F-111 horizontal stabilizer program was initiated to make a more extensive application of filament composites. This stabilizer utilizes a full depth honeycomb sandwich with boron epoxy skins with 0° ± 45° fiber orientation. The weight saving in this application amounted to 25 per cent overall but to 60 per cent for the boron members only.

In addition to these laboratory test programs, functional flight test programs are currently being performed on:

F-111	Wing Trailing Edge Panel	
	Aluminum	16.3 pounds
	Boron	14.0 pounds
	Sandwich Structure with boron face sheets with 90° ± 30° orientation	
	Flown for 2 years with no visible degradation	
F-111	Landing Gear Door	
	Aluminum	20.8 pounds
	Boron	17.3 pounds
	Sandwich Structure with boron face sheets	
	Flown for 2 years with no visible degradation	
F-4	Rudder	
	Aluminum	64.0 pounds
	Boron	40.0 pounds
	Sandwich Structure with boron face sheets with 0° ± 45° orientation (Torsional critical)	
A-4 Flap	Aluminum	21.6 pounds
	Boron	16.9 pounds
	Carbon	13.2 pounds
	Sandwich Structure with boron face sheets 0°, 90° ± 45° orientation, (Deflection critical)	
F-5	F-5 Landing Gear Door	
	Aluminum	17.5 pounds
	Boron	12.4 pounds
	Sandwich Structure boron face sheets 0°, 90° ± 45° orientation, (Stiffness critical)	
F-5	Leading Edge	
	Carbon	

Except for the new F-111 horizontal stabilizer program weight saving was not the motivating factor behind these programs. The primary goal had been, as mentioned previously, to gain engineering design data, to develop hardware, to establish manufacturing techniques and machinery and to instill confidence in these materials. That these goals have been attained can be attested to by the fact that boron epoxy structures are being developed for: the C-5A leading edge slats, the F-4 rudder and the F-14 horizontal stabilizer.

5. DESIGN CONCEPTS

In order to improve the performance of the structural materials available the designer has

developed means for more effectively utilizing the mechanical properties of the materials by improving joining and attachment techniques and methods, by utilizing cross sectional shapes which more efficiently utilize the material employed or by utilizing materials with different properties in such a manner as to exploit only those outstanding qualities desired.

Currently, design concepts which show promise for the future are the extensive use of welding for joining and attaching, particularly the use of electron beam welding which has little effect upon the material's properties and which can weld very thick sections in a single pass; the use of diffusion bonding which offers even greater advantages than welding as there is no degradation of the material's properties, there is no need for excess material nor space required for joining and the pieces to be joined may be so oriented as to take full advantage of their best properties (no grain flow or low short transverse properties for forgings); and the use of laminated structural members in which thin gage, high strength sheets are bonded together to form not only a much stronger unit but one which has greater resistance to crack propagation and sonic fatigue.

6. MATERIAL INFLUENCES

The major direct influence that a new material can exercise on aircraft design is that of saving weight in the basic airframe. A saving that may then be reflected as improved performance - increased range, higher cruise altitude, improved take-off and landing characteristics - greater payload or a smaller aircraft to accomplish a given mission.

A preliminary estimation of the weight saving potential of a new material can be made by a weight merit factor which consists of the ratio of that specific mechanical property - mechanical property divided by the density - for the loading state being evaluated for the base material to the new material.

Examples of these weight merit factor parameters are: F_{tu}/ρ tension, F_{cy}/ρ local buckling, $E^{1/2}/\rho$ general instability, $E^{1/3}/\rho$ lateral deflection or shear buckling where: F_{tu} is the ultimate tensile stress, F_{cy} is the compressive yield stress, E is Young's Modulus and ρ is the density.

Table 2 (10, 11) contains a listing of the specific properties for representative structural materials at room temperature. An item to note is that no one material is outstanding in all factors.

This method of estimating potential weight savings is for those instances in which the load is transmitted over the same distance by geometrically similar structures.

In conducting these influence studies it is necessary to consider not only the already delineated strength and stability factors but also the potential gains or losses due to fatigue, stress corrosion, fracture toughness, joining effectiveness and minimum gage requirements.

Figure 13 (10, 11) is a plot of specific strength - F_{tu}/ρ - for several candidate materials plotted against temperature. Also noted on the ordinate are the Mach numbers which generate lower surface equilibrium temperatures of those magnitudes. The Mach numbers and the associated temperatures are plotted in this case to emphasize the fact that, for high speed aircraft, once the structural material is selected the upper limit of the aircraft's speed capability is established.

An item of even greater note, however, is the relatively low position of aluminum - the current basic material of aircraft construction - in comparison to steel and titanium. A fact that can be explained by cost, acceptance, machine tool availability and the presence of a large labor pool. All of which mitigate against a change in materials except for those components such as engine bays, pylons, cowlings, and aircraft when the operational temperature is above the limit for the successful application of aluminum. In most of these instances titanium has been successfully employed to not only make the component or airplane possible but to also provide a weight savings of 10 to 20 percent.

Beryllium offers even greater potential and, in fact, is the best of all the metals. And when the specific modulus (E/ρ) is considered the margin is even greater. Beryllium hasn't seen wider structural application other than in the Minuteman interstage, Agena and C-5A brakes because of a lack of ductility, its toxicity, high cost and lack of confidence in its suitability.

For those failure modes which are not directly associated with strength or for resizing purposes, it is necessary to compare the materials or structural design concepts by means of a structural index which characterizes the relationship between strength and size that is common to an entire family of structures obtained by the dimensional similitude transformation.

An example of material and design concept effectiveness in weight saving is shown in Figure 14 (12).

Figure 14 (12) indicates the potential reduction in the take-off gross weight versus the percentage of titanium utilization in the structure for a fixed size aircraft and a resized aircraft in which the original performance functions of wing loading, thrust to weight ratio and fuel fraction were held constant.

The materials with the greatest potential for structural weight saving and consequently the greatest potential for performance improvement are the filament composites. For, in addition to

their great specific strengths and stiffness, they possess the capacity of being "tailored" to ideally transmit the required loadings with maximum efficiency. It is these materials which will also present to the designer the greatest of all challenges because their outstanding potential for weight saving will require more precise definition of all loading conditions, an increased understanding of their engineering characteristics and an increase in the ability of the designer to "tailor" the material to the design conditions.

For a combat aircraft where even a small performance improvement can well make the difference between having a positive win/loss ration because of the "S" shaped curve of military effectiveness versus performance and where the growth factor is in the order of ten to one, the effective application of new materials is essential.

7. PROBLEMS IN THE EARLY APPLICATION OF NEW MATERIALS

In the preceding sections the trends in aircraft operation and design were noted and discussed, the developments in new and improved materials were highlighted and the potential improvements in performance were outlined. It must be noted, however, that the introduction of new and/or improved materials - Figure 6 - into aircraft design requires an inordinately long period of time, if, indeed, it is accomplished at all.

Associated with improved performance is the requirement for additional materials data and more data particularly as related to real time data for supersonic and hypersonic aircraft or even long life subsonic aircraft.

Why in this era of increased emphasis on and formalized tools for management to reduce the time required to develop aircraft does such a situation exist?

Explanations range from too costly to lack of confidence, inadequate or inexperienced labor pool, limited production sources or suppliers, or offsetting materials characteristics. All of which are true and reasonable, to a degree. Moreover, the situation itself - the greatly reduced development time and the warrantee requirements within the contract - forces the designer to stay with proven state-of-the-art materials.

Another and equally important reason does exist, however, in the form of a lack of communication and understanding between the materials specialists and the designer. As a result, new alloys are developed or certain mechanic properties of a material are improved for a specific application without a thorough understanding of the critical parameters with the ultimate result being the failure of the part or component in service. For example, the emphasis on improved ultimate tensile strength without regard to fatigue or the high strength forging materials with high susceptibility to stress corrosion.

Problems such as these, in conjunction with other service failures and the demand for increased reliability have resulted in the designer placing so many and so stringent demands upon the materials specialists that it is virtually impossible to obtain all of the required data in the 10 to 15 years required to introduce a new material into the production cycle much less in the 4 year aircraft development time.

8. CONCLUSIONS

It is concluded that there are many materials available which, if applied, could exert a considerable influence on the performance of a military aircraft but that they are being withheld for numerous reasons, the most likely being that of cost effectiveness.

The only solution to this dilemma is the improvement of communications and a better understanding of these requirements between the materials specialist and the designer who will then set up a program, modest in scope but sufficient in detail, to enable a new material to transition from a laboratory curiosity to development hardware to aircraft components with the proper data in the correct amount being obtained for each phase of the development. To guide both the designer in his quest for materials with improved properties mission analysis/parametric trade studies should be made to determine those mechanical and physical properties of the materials which, if they can be improved, would have the greatest potential payoff and their physical limitations.

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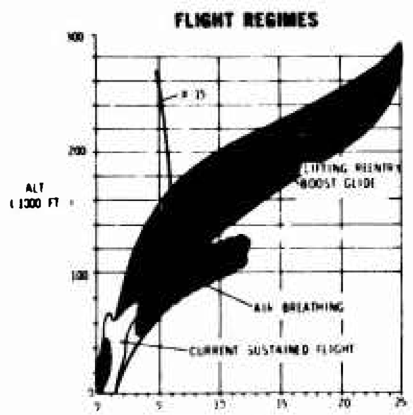


FIG. 1

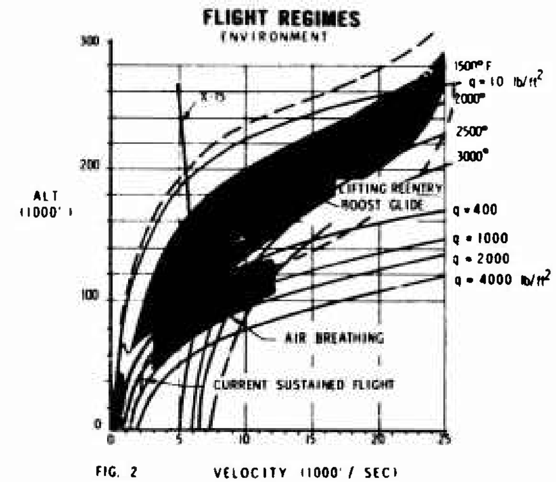


FIG. 2

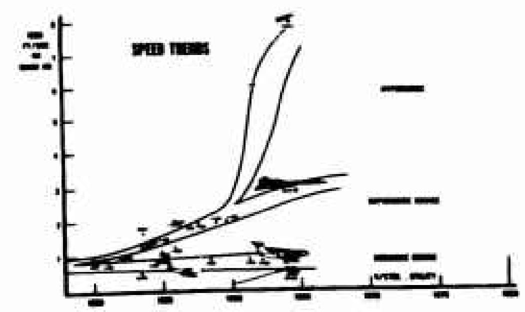


Fig 3

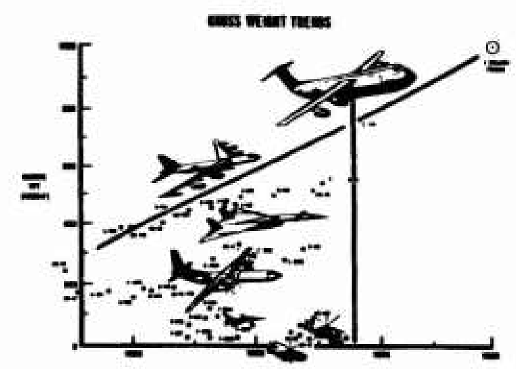
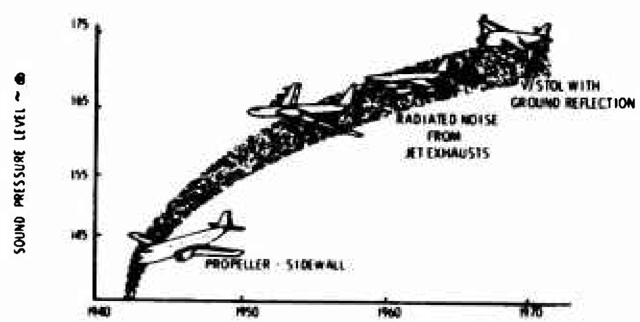


Fig 4

TRENDS OF AIRCRAFT NOISE



YEARS
FIGURE 5

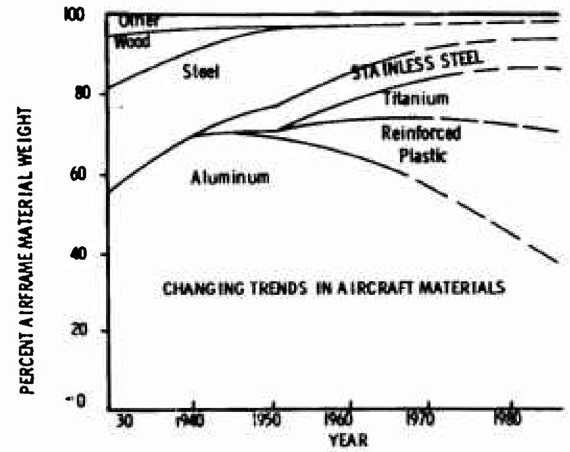


FIGURE 6

STRUCTURAL WEIGHT FRACTION (λ_s) TRENDS

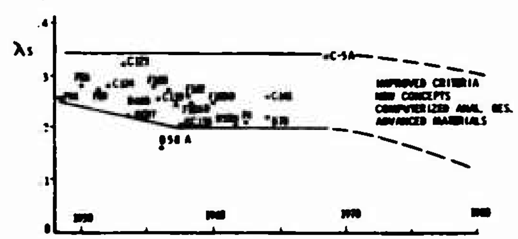


Fig 7

MATERIALS USAGE

F-4

GROUP	TOTAL WT.	AL.	STEEL	TI.	MAG.	LEAD	GLASS & PLASTIC	RUBBER
WING	[illegible]	[illegible]	[illegible]	[illegible]	7	0	0	0
TAIL	[illegible]	[illegible]	[illegible]	[illegible]	6	[illegible]	11	0
FUSELAGE	[illegible]	[illegible]	[illegible]	[illegible]	47	0	314	27
LANDING GEAR	[illegible]	[illegible]	[illegible]	0	0	0	1	137
ARRESTING GEAR	[illegible]	0	[illegible]		0	0	0	0
TOTAL	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]
%	100	67.25	[illegible]	[illegible]	[illegible]	0.22	2.57	1.30

FABRICATION PROCESS

SHEET METAL	[illegible]	[illegible]	20.22	6.91	[illegible]	0	0	0
MACHINED	[illegible]	[illegible]	[illegible]	[illegible]	0.37	[illegible]	0.50	0
MOLDING & SHAPING	3.57	0	0	0	0	0.20	2.70	1.30
	100	67.25	[illegible]	7.35	0.47	0.22	2.57	1.30

FIG. 8

Material Usage Rating

MATERIAL	USAGE	FORM-ABILITY (1)	MACHINE-ABILITY (1)	WELD-ABILITY (1)	AVAILABILITY (2) 1969	AVAILABILITY (2) EST 1975 1980	MATERIAL COST (3) 1969	MATERIAL COST (3) EST. 1975 1980
ALUMINUM	CREW AREA AND STRUCTURE TO 250°F	A	A	C	A	A	A	A
	EQUIPMENT AREA AND STRUCTURE TO 350°F	A	A	C	A	A	A	A
	CRYOGENIC AND LOW PRESSURE TANKS, VEHICLE	A	A	B	B	A	A	A
TITANIUM	CRYOGENIC TANKS (ELI GRADE) SHEET, PLATE, BAR, FORGING	B	B	A	A	A	B	A
	SHEET METAL STRUCTURE TO 1000°F - (ANNEALED)	B	B	A	B	A	B	A
	STRUCTURE-SHEET, PLATE, BAR, FORGING TO 800°F (H.T.)	B	C	C	B	A	B	A
	STRUCTURE-FORGINGS OVER 4" SECTION - (H.T.)	-	C	D	B	A	B	A
INCONEL	STRUCTURE TO 1400°F - SHEET, PLATE, BAR, FORGING	B	B	B	A	A	A	A
RENE'	STRUCTURE, HEAT SHIELDS, SHINGLES TO 1500°F SHEET, PLATE, BAR, FORGINGS, FASTENERS	B	B	C	A	A	B	B
L - 605	STRUCTURE, TO 1800°F SHEET, PLATE, BAR, FORGINGS, FASTENERS	B	B	B	B	A	B	B
T.D. NICKEL	STRUCTURE, HEAT SHIELDS, SHINGLES - TO 2400°F SHEET, BAR	A	A	D	B	A	C	B
COLUMBIUM - D-43	STRUCTURES TO 3000°F - MUST BE COATED	B	B	C	B	A	C	B
	SHEET, PLATE, BAR	A	B	B	B	A	C	B
	Fs-85 AND SCb - 291 USED WHERE HIGH	A	B	B	B	A	C	B
	CREEP RESISTANCE AT HIGH TEMPERATURE IS REQUIRED	A	B	B	B	A	C	B
MOLYBDENUM - MO	STRUCTURES TO 3000°F - MUST BE COATED	C	C	D	B	A	C	B
	SHEET, PLATE, BAR	C	C	D	B	A	C	B
	HIGHER STRENGTH THAN Cb ALLOYS AT 3000°F	C	C	D	B	A	C	B
BERYLLIUM - Be-2.0 BeO	SHEET, PLATE	C	B	D	B	A	C	C
	BAR, BILLETS	C	B	D	B	A	C	B
	FORGINGS	C	B	D	B	A	D	C
	SHEET, PLATE	B	B	C	C	A	D	B

(1) FABRICATABILITY
(A) READILY
(B) MODERATELY DIFFICULT
(C) VERY DIFFICULT
(D) CONSIDERED IMPRACTICAL

(2) AVAILABILITY
(A) LARGE VOLUME PRODUCTION
(B) LOW VOLUME PRODUCTION
(C) EXPERIMENTAL & PILOT PROD.
(D) IN RESEARCH & DEVELOPMENT

(3) COST
(A) LOW-LESS THAN \$5/LB
(B) MEDIUM-\$5 TO \$25/LB
(C) HIGH-\$25 TO \$250/LB
(D) VERY HIGH-OVER \$250/LB.

TABLE - I

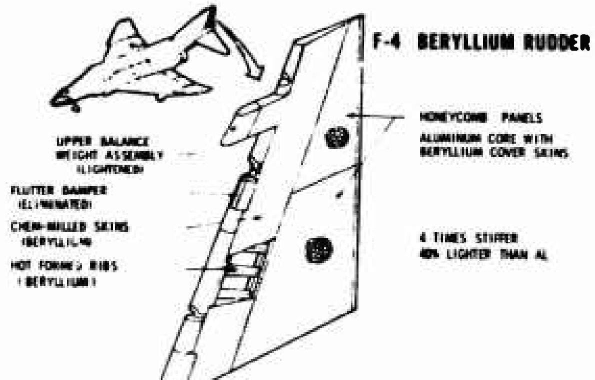
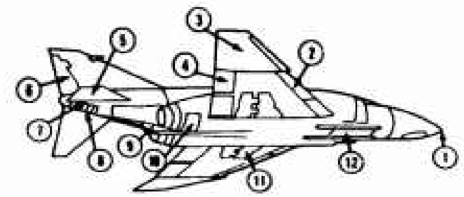


FIGURE 9

Beryllium Application



F-4 COMPONENT	PRODUCTION VERSION MATERIAL	PRODUCTION VERSION WT. (LB.)	BERYLLIUM VERSION WT. (LB.)
1. RADAR ANTENNA SUPPORT	AL	29	12
2. LEADING-EDGE FLAPS	AL	277	179
3. OUTER WINGS	AL	305	229
4. AILERONS	AL, STEEL	100	63
5. STABILATORS	STEEL, TI, AL	641	385
6. RUDDER	AL	64	42
7. STABILATOR ACTUATORS	STEEL	38	25
8. TAIL CONE PANEL	TI	64	43
9. ENGINE BLAST FAIRINGS	STEEL	66	40
10. ENGINE ACCESS DOORS	AL, TI	431	297
11. MAIN LANDING GEAR BRAKES	STEEL	196	144
12. NOSE LANDING GEAR DOORS	AL	29	19
	TOTAL	WT. 2246	1478

FIGURE 10

MANUFACTURING BREAKDOWN

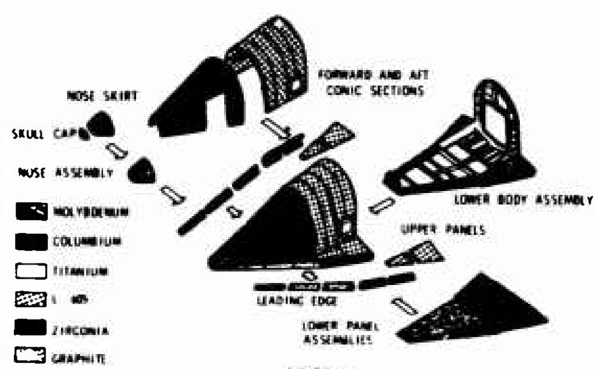


FIGURE 11

HYPERSONIC AIRCRAFT STRUCTURE

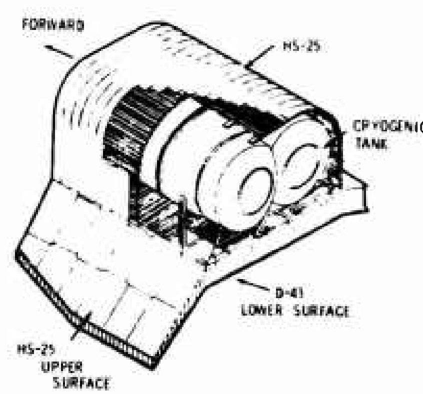


FIGURE 12

MATERIAL SPECIFICS									
MATERIAL	F_{tu} Ksi	F_{cy} Ksi	E Msi	ρ lbs/in^3	F_{tu}/ρ in x 10^5	F_{cy}/ρ in x 10^5	E/ρ in x 10^5	$E^{1/2}/\rho$ in^2/lbs $1/2 \times 10^3$	$E^{1/3}/\rho$ in$^{7/3}$ lbs$^{2/3} \times 10^3$
7075-T6	82	72	10.5	.101	8.14	7.14	1040	32.10	2.16
Ti-6 AL-4V	162	157	16.4	.160	10.12	9.82	1024	25.30	1.58
PH15-7MO	240	240	32.1	.277	8.67	8.67	1160	20.40	1.15
BERYLLIUM	76	60	42.8	.067	11.35	8.96	6380	97.80	5.20
Cb-752	95	80	15.0	.326	2.92	2.46	460	11.80	0.75
TUNGSTEN	170	-	52.0	.697	2.44	-	746	10.30	-
BORON (1)	200	200	32.0	.072	27.80	38.80	4450	81.20	4.40
CARBON (1)	150	70	25.0	.052	28.90	15.20	4810	96.10	5.60
S GLASS (1)	210	80	7.6	.073	28.80	10.95	1040	37.80	2.70

(1) UNIDIRECTIONAL FIBERS 50% BY VOLUME IN EPOXY

TABLE 2

SPECIFIC STRENGTH

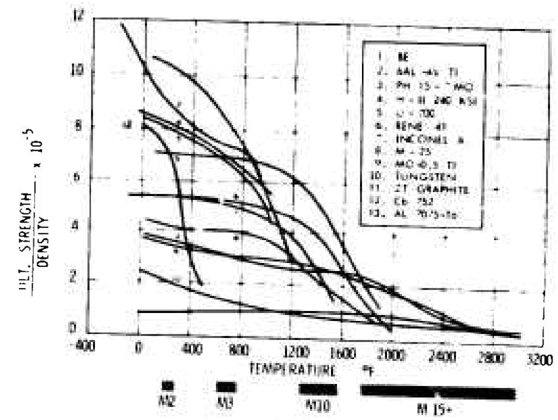


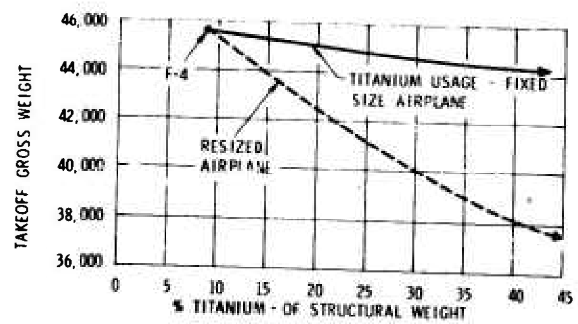
FIG. 13

COMPRESSION STRUCTURE COMPARISONS

DIFFUSION BOND

TYPE	PLATE	SKIN-STRINGER	HONEYCOMB	ROLL BOND
CONFIGURATION	10.0 in	3.0 in	0.375	0.375
COMPRESSION LOAD (lb/in)	16,000	16,000	16,000	16,000
WEIGHT (lb/sq ft) TITANIUM	5.7	4.6	3.3	3.0
WEIGHT (lb/sq ft) ALUMINUM	4.7	4.4	3.8	

FIG. 14



VARIATION OF TAKEOFF GROSS WEIGHT VS PERCENT TITANIUM STRUCTURE

FIG. 15

AIRFRAME SYSTEMS DESIGN EVALUATION

by

Lawrence P. GREENE

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AIRFRAME SYSTEMS DESIGN EVALUATION

by

LAWRENCE P. GREENE

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SUMMARY

A general view of the compromises a designer must make in aircraft subsystems to achieve the correct and optimum effectiveness of the total aircraft system is presented. The emphasis is directed at resolution of problems relating to flight mechanics functions. It is intended to identify the contribution an imaginative systems designer can make to an effective flying machine, which in turn is recognized to be one of the necessary assets of military aircraft.

The essence of any successful design is the understanding the proponents of each associated discipline exercises when dealing with others of the design team. Examples of design ingenuity are shown to illustrate some of the areas where innovation provided reasonable answers to otherwise troublesome problems. General observations of the requirements imposed on supporting airframe systems are developed, with particular attention to control system options, power sources, environmental problems, and total system dependability.

INTRODUCTION

For Aeronautical Generations, the old cliches about a designer's dream vehicle has been a joke with a very pointed meaning. The importance of "true" optimization is upon us and optimization takes different forms. Now, it is several talents, several disciplines working together to produce a really useful machine. Aeronautical engineers used to get locked into their own discipline -- then gradually they began to realize that the old quote about "if you put enough power on it, you can fly a barn door" was not enough. It is no longer enough to just fly, the job now is to fly well, to develop the best compromise.

A military preliminary designer today knows that he must exercise many tradeoffs between the "best of one disciple and the best combination. Today, it must be worth it. It is the intention of the author to identify examples of the values that are realized in making a good design a great design. Many times the success or failure of a design rests in the hands of a man who is an imaginative, innovative designer, without whom, the great "breakthrough" in scientific application will be a hollow thesis. He may not be a scientist but it is his ingenuity that makes the idea live.

DISCUSSION

The great designs, however, are the culmination of a set of challenges that produce something that is not just "old hat". It is "way out". That great design is not the greatest power plant, aerodynamic design, structure or system but the greatest combination. It is the product of a group that surpass themselves and produce something that is just a little, but an important, bit better than their competition.

Some of the most demanding of design trades are involved in the application of the "same old stuff". Here in the mechanical, electrical, and fluid power systems are the frequently unsung innovative efforts which contribute prominantly to a great flying machine. The design conditions are often demanding, the operating environment challenging, and here are the opportunities for system failure, for poor serviceability, for missing peak performance.

The trades are well recognized; weight or simplicity versus performance, serviceability and accessibility versus performance, complexity and redundancy versus safety, etc. The preliminary designer trying to get the maximum out of his particular combination of payload/range requirements, speed/altitude/acceleration spectrum calls upon the system designer for help.

An exotic aircraft of a few years ago was designed to explore the unknown of speeds of 4,000 miles an hour at altitudes that qualified its pilots as astronauts (over 50 miles high). For the complexity of the system, the preliminary design specification was a masterpiece of simplicity. (Figure 1) The vehicles objective was to explore the unknown, achieve a speed of Mach 6, reach an altitude of 250,000 feet and make the structure tolerate a skin temperature of 1200°F. Except for the subjective quality of the unknown which pervaded everything, no two of these requirements could be achieved simultaneously. It was recognized during the preliminary design that every discipline would be stretched to the maximum.

The vertical tail configuration was an example of several component design requirements being met with a unique combination. The directional stability requirement was most difficult to meet at the high Mach numbers; very effective speed brakes were required for speed control. Directional control was required to be excellent at all Mach numbers and minimum weight was required. A relieving factor was that drag was relatively unimportant for the mission.

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These combined requirements resulted in the vertical tail configuration shown on Figure 1. The airfoil section was a 10 degree wedge. The wedge with its blunt aft end would not have been acceptable if drag had been an important consideration. However, the wedge section does increase the surface lift curve slope at hypersonic Mach numbers, just where the vertical stabilizer contribution to directional stability was needed most. Figure 2 shows the directional stability parameter C_{n_β} with the vertical tail off, with a diamond shaped vertical stabilizer, and with a wedge shaped vertical stabilizer, as measured in early wind tunnel tests. (As a matter of interest, the final configuration did exhibit positive levels of C_{n_β} throughout its Mach number range rather than neutral stability shown for this early wind tunnel model.) It can be seen that the wedge airfoil section did provide a significant increase in the vertical stabilizer contribution to C_{n_β}. A similar increase in vertical stabilizer effectiveness, $\Delta C_{n_{\delta_v}}$, was also realized.

Figure 2 also shows an increase in directional stability when the speed brakes were opened. This was a desirable feature, in that it provided a readily available means of increasing directional stability, should it be desired. The speed brakes in this location also provided the required drag increase throughout the airplane's flight envelope, and a minimum input to the airplane's longitudinal characteristics.

The configuration was attractive only when a satisfactory structural arrangement was found. Figure 3 shows the general arrangement. The wedge airfoil section provided adequate space for the movable stabilizers torque tube, and the speed brake actuator was located within the wedge section, aft of the torque tube. Thus an unusual vertical tail configuration was evolved to meet the peculiar requirements of the world's first hypersonic airplane. But in doing so, it put some real demands on the hydraulic system.

The hydraulic system consists of two completely separate, airless, modified type III, 3,000 psi systems operating in parallel. Only the flight control surfaces are driven hydraulically, as there is no utility system. The operating temperature range of the equipment, lines, and fittings is from -65°F to 400°F. The main problems encountered in the design of this system are due to the extreme temperature and vibration conditions. These problems made it necessary to find a new hydraulic fluid, new seals, new materials, better methods of fabrication, installation, and contamination control, and tighter "quality" controls.

In order to establish temperature requirements for the hydraulic equipment, an analysis, based on the high-altitude maximum-speed mission which appeared to cause the most severe heat problem, was made. It was calculated that if the temperature of the hydraulic fluid was maintained at approximately -20°F at take-off time (Fig. 4), it would rise to 0°F during captive flight, reach 50°F during the five minute warmup period, and 300°F to 400°F during the last eight minutes of free flight. In addition, there would be an estimated 22 minutes of soak after landing. For 15 flights of this type, the total time that the system would be at or near 400°F would be about 7.5 hours. All the hydraulic equipment was designed on the basis of these data, together with low temperature (-65°F) and proper transient-condition considerations. Considerably more hours of usage are available at lower temperatures as determined by testing.

Many fluids were considered and those most likely to meet the requirements were tested, and comparisons made. Viscosity at 400°F was considered very critical in that this property, if too low, would reduce the volumetric efficiency of the pump and also would allow for excessive leakage in our valves, so that increased pump output would be required, and this in turn would cause system fluid temperatures to be elevated. Lubricity was considered a very important factor in that the operation of the hydraulic pump, especially at 400°F, is the "heart" of the vehicle.

This is an example of a very complex machine that set the pace for some of the later military designs. This was a transient aircraft designed for a single purpose. It is a tribute to the system designers that the aircraft were modified and adapted, extended and flown for longer than anyone had expected. The systems were dependable.

Let us examine another unique approach. We tried to optimize a vehicle for a long duration operation in a very hostile environment. There were many detail changes between preliminary design and final construction and operation but the basic design was maintained, accomplished and proven satisfactory. The speed altitude diagram is shown in Figure 5 with a time sequenced set of arrows that shows the speed altitude combinations maintained throughout the mission. Ninety percent of the distance covered on the design mission was to be done at M=3, this required about 70 percent of the time of the mission (about 2.5 hours). Structural temperature stabilization was achieved in about 15 minutes at M=3. The combination of temperature, altitude, Mach number, maneuvering requirement, size and weight of the airplane summarized the flight mechanist's interests but that only described the starting conditions on the systems design. Figure 6 illustrates the primary novel configuration features of the airplane; I call attention to the two position windshield first. The aerodynamic design required a high cruise efficiency at the design cruise speed of 3.0 Mach number. To optimize the aerodynamic efficiency, as represented by the cruise lift-drag ratio (L/D), it was found that a smooth faired canopy line would result in a significant reduction in drag. The smooth faired canopy did not provide the necessary pilot visibility for take-off and landing, therefore, a movable ramp canopy was designed. Drag tests indicated that at subsonic speeds, there is no drag differences between the two canopy configurations (Figure 7). At supersonic speeds, the difference in drag amounts to a change in maximum (L/D) of .25.

The windshield consists of two transparent subsystems, providing forward, upward, and side vision for the pilot and co-pilot (see Figure 8). The outer glass subsystem is part of the air vehicle

mold line, forming a continuous, unbroken contour during high speed flight. For maximum vision during landing, take-off, or in-flight refueling, the outer windshield may be depressed. An air gap separates the outer glass assembly from the inner windshield subsystem which is fixed to the structure, forming part of the cabin pressure barrier.

The outer windshield is composed of five monolithic full-tempered glass panels, each with its own titanium alloy frame. Each panel, mounted in its frame, forms an inter-changeable unit.

The movable windshield assembly is sealed to the fixed structure by a metal bellows at the aft end and silicone rubber impregnated fabric seals along the sides and front. These seals prevent dust and moisture from entering the gap between the inner and outer glasses. However, should the glass require cleaning on the inside of the gap, the upper outboard panels may be hinged open for access.

Vision through the windshield under all extreme climatic conditions is implemented by three air vehicle subsystems. A hot air de-icing and rain removal system clears the outer surface of the movable windshield. A filtering and drying system prevents contaminants from entering the air gap between the inner and outer windshield and depositing on the glass surfaces. An electrical defogging system heats the inner windshield glasses, preventing moisture from collecting on the inner surfaces.

Going back to Figure 6, I wish to call attention to the folding wing tips. The valve of this concept to the flight mechanist was three fold: Improved directional stability (Figure 9), improved long stability characteristics (Figure 10) which in turn decreased the trim characteristics required and reduced drag. With these design features incorporated, the airplane was capable of being flown efficiently and safely without stability augmentation systems at all speeds. SAS was included for optimized flight characteristics but was not required for safe operation. These capabilities were achieved through the efforts of some highly ingenious mechanical and hydraulic design contributions.

In the case of this airplane, the mission profile took it into a hostile temperature environment that required development of entirely new approaches to operational equipment. The stagnation temperature of up to 630°F at Mach 3 with the resulting high temperature for critical systems aboard eliminated much of the conventional hardware available. Innovation was necessary to insure the presence of components which could perform compatibly in this higher temperature regime. These points are illustrated on Figures 11, 12 and 13, and the technical aspects summarized here.

Wing Fold Data

1.	Hinge Material	= H-11 steel
2.	Operating temperature	= -65 to 600°F
3.	Hydraulic Motor Drive (two units per side)	= 380 in-lbs @ 4000 psi, (static) 340 in-lbs @ 3500 psi a 5500 RPM
4.	Speed Reducer Ratio	= 12.552 to 1
5.	Hinge Ratio	= 2534.1 to 1
6.	Total Gearing Ratio	= 31,812 to 1
7.	Hinge Shaft Speed	= 438 RPM
8.	Wing Fold Ratio	= 1.04°/sec
9.	Hinge Holding Torque	= 4.08×10^6 in-lbs/unit
10.	Hinge Operating Torque	= $.67 \times 10^6$ in-lbs/unit
11.	Hinge Stall Torque	= 2.4×10^6 in-lbs/unit
12.	6 Hinges are used per wing	
13.	Stiffness/Hinge	= 195×10^6 in-lbs/radian @ 0° position = 169×10^6 in-lbs/radian @ 25° position = 105×10^6 in-lbs/radian @ 65° position
14.	Weight/Hinge	= 193 lbs
15.	Hinge Efficiency	= 47%

An example of the importance of a full system approach to the solution of an operating problem can be seen in the design of the fuel subsystem (Figure 14). The high structural temperatures from aerodynamic heating would raise the temperature of the fuel contained in the integral tanks. The demands on the hydraulic system required operating pressure of 4,000 psi which necessitated a cooling system for the fluid. To save space, the hydraulic lines were routed through the fuel. The environment control system also had to dissipate heat. Because the fuel would be consumed, it was a logical medium of heat exchange. (Figure 15) Therefore, the fuel system provided a series of heat exchangers for the various supporting systems, but the arrangement and capacity of these heat exchangers had to be such that the resulting temperature of the fuel entering the engine would not exceed that which would deteriorate the fuel quality to the point where it would affect engine performance, vaporize, or cause coking. Consequently, the fuel system was designed to operate at a high pressure level to avoid fuel boiling, the supporting systems heat exchangers were sized and arranged to deliver fuel to the engines at a maximum of 300°F, and the pumps, valves and controls were designed of new materials and seals to perform at 300°F instead of the former 160°F temperature. (Figure 16) This situation demanded the total system approach to the problems (and the requirements) and entailed intimate coordination between the fuel system designers and the engine manufacturers, the hydraulics engineers, the structural engineers, the environment control engineers and the petroleum industry.

Similarly operation of the airplane in a high temperature environment posed a problem never before satisfactorily solved, that is, prevention of explosion of the fuel vapors in the tanks due to auto-ignition as the vapor temperatures reached 430°F. This could result from the high structural temperature anticipated at Mach 3 when the tanks were empty. This situation required the development first of a concept which would prevent ignition and second of a hardware system that would implement that concept. The highly successful fuel tank inerting system utilizing nitrogen to reduce the oxygen concentration in the fuel tank vapor space to a level which would not support destructive combustion was the result. This system development required careful analysis and testing to determine performance requirements. Then a completely integrated system of components was developed to accomplish the required performance. This system had to perform with complete assurance that the fuel vapors were immune to explosive concentration and in addition it was utilized to invoke a tank pressurization schedule to reduce fuel losses from boiling. Because of these high pressures, it was difficult to seal the tanks to contain nitrogen. It can be seen that the design approach in this critical application had to be such that 100% reliability was assured.

Now let's talk about a "simple airplane". Here (Figure 17) is a two-place, twin turboprop airplane designed primarily for armed and visual reconnaissance, helicopter escort and forward air control and secondarily, to deliver troops and provide logistics support. It is designed to operate out of short, unprepared rough fields with a minimum of maintenance and support equipment. The basic configuration is shown in this three-view chart (Figure 17). The weight empty is 6,969 pounds and the airplane is designed for a load factor of 8 at 9,390 pounds. The design landing sink rate is 18.8 feet/second at 10,044 pounds on rough fields and the maximum overload takeoff weight is 14,466 pounds (more about this later). The flight control systems are manually operated and aerodynamically balanced. The lateral control system consists of ailerons with both geared and spring tabs with rotary plate spoilers located ahead of the outboard flaps. The spoilers are mechanically connected to the aileron through a mechanism which varies the spoiler travel with airspeed. The rudders are aerodynamically balanced and do not have tabs. Force trim is utilized on all three axes. Four equal span, double slotted flaps provide additional lift for takeoff, landing and low speed maneuvering.

The design flight envelope of pressure altitude versus equivalent airspeed is shown in Figure 18. The significant features are: The 430 knot limit speed from sea level to 4,000 feet, where the maximum permissible Mach number is reached, the 220 knot average cruise speed at normal rated power at sea level, the 120 knot average loiter speed at sea level and the 70 knot landing speed at 8,500 pounds gross weight.

This is the airspeed/load factors diagram for the design combat weight of 9,390 pounds at 4,000 feet. Shown are the positive 8 and negative 3 "g" limits for symmetrical maneuvers and the 6.4 "g" limit for rolling pull-outs. The buffet boundary and dynamic stall regions are shown on the left.

With this general background of the OV-10A presented, let us now review the control system development (Figure 19).

In order to provide the lifting capability required to allow operation at the low speeds necessary for short takeoff and landing, a powerful high lift device was required. In addition, placing a large percentage of the wing span in the propeller wash produced significant lift due to power and as one might suspect, provided many problems.

Slotted elevators were incorporated to provide effectiveness for angles up to the 35 degree trailing edge up. Elevator hinge moment control is provided by aerodynamic balance and trailing edge tabs. In keeping with the design goal of minimum maintenance, an unboosted mechanical system was required. A spring tab system was selected because of the ability of this type of system to compensate more completely for the widely varying requirements over the complete Mach number and dynamic pressure range.

The pilot's stick is connected to the spring tabs through a series of push rods and a cable-crank system. Trim is provided by an electrically actuated torsion bar which mechanically relieves the pilot force. The design requirements for maneuvering stick force gradients dictated extremely close hinge moment balance; consequently, rather large tab gearing ratios were necessary. The system provides boost ratios of about 10, thus producing the same trick force reduction as a representative boosted system. All of this merely illustrates that the challenge to a control system designer was for innovation, imagination to the same degree as required for the more sophisticated aircraft.

The airplane was also required to operate at low altitudes and at all flight speeds in a hostile environment of ground fire and would be subject to extensive battle damage. To further increase the longitudinal control system reliability, the control system cables, bellcranks and push-pull rods were run from the fuselage through each boom to separate the system. Severance of the actuation through one boom still allows control by the remaining side for only a modest increase in control force.

A similar concept was used on the lateral control system. Severance of a single control system cable still allows a high degree of lateral control. In addition, the plate spoilers operated normal to the wing surface and airflow produce rolling moment with essentially no hinge moment.

The directional control system did not require tabs for force reduction. The redundancy in this system is inherent with the twin vertical tails.

The definition associated with unprepared airstrips required the airplane to have good flotation, the ability to land on bumps, steps and holes, and to taxi over series of undulating contours. At the same time, it was clear that fulfillment of the specification requirements would present a real challenge in the design and development of its landing gear and maintain a rugged structural design.

The requirement that the airplane be able to operate off of rough terrain and unprepared airstrips was of primary importance in the design of the landing gears. Since these operations showed such wide variation in directions of applied loads that gear stroking difficulties could be anticipated for the conventional telescoping gear configuration. Because of the favorable drag-vertical load relationship, the main gear is articulated and includes a neutral oleo assembly mounted on right and left hand post, axle beam and drag brace assemblies. The nose gear uses a levered geometry known as semi-articulated because of the torque link to the axle beam permitting a fixed center for full 360° castering.

In conclusion, it was intended to illustrate the extensive and important contribution of the airframe subsystems designer in adjusting to the demands put on him by the importance of the military mission. The flight mechanics characteristics are the things the military operator feels. What he doesn't see and frequently doesn't appreciate is the ingenuity that has been used to achieve any capability at all. The greatest theories of aerodynamic efficiency, the wildest dreams, which begin with "If only . . ." would be nightmares if it were not for the creativity of the designer. I leave you with the thought ". . . exceptional design involves exceptional ingenuity".

ACKNOWLEDGEMENT

In preparing this paper, the author is grateful to and acknowledges the support of Messrs. E. W. Johnston, R. Crone, R. Lyford, Keith Hayden, and their staff, of North American Rockwell Corporation, in preparing and submitting the information.

FIG. 1 **X-15 SPEED BRAKES**

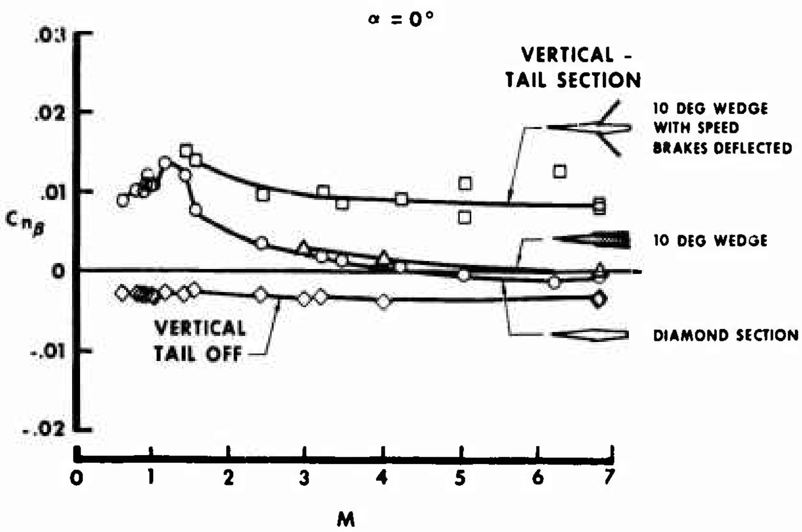


FIG. 2 **DIRECTIONAL STABILITY**

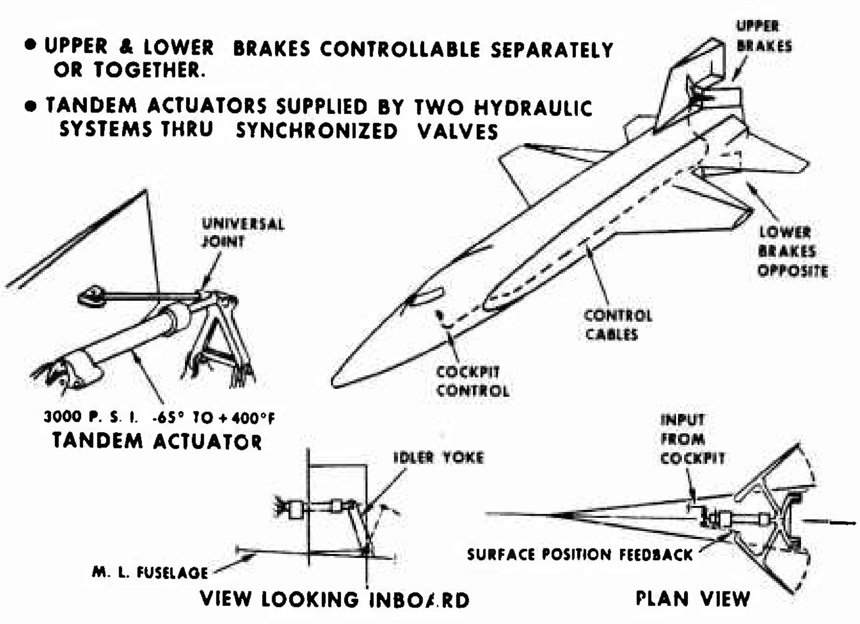


FIG. 3 **X-15 SPEED BRAKES**

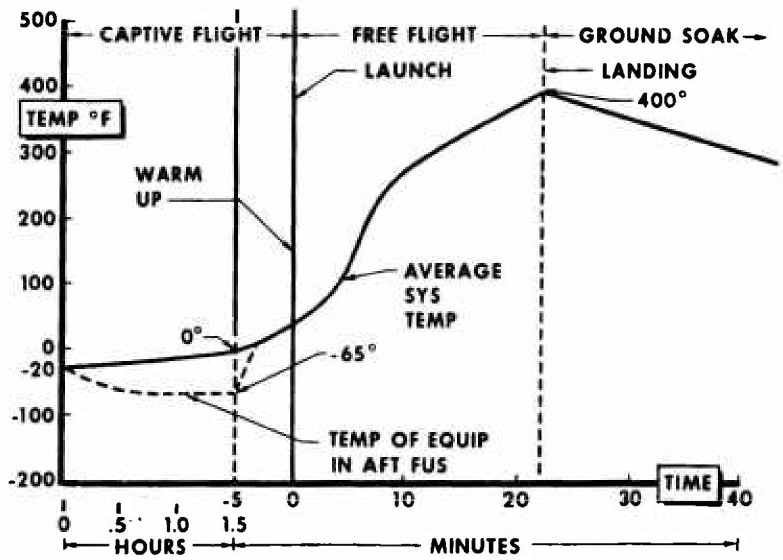


FIG. 4 TEMPERATURE PROFILE

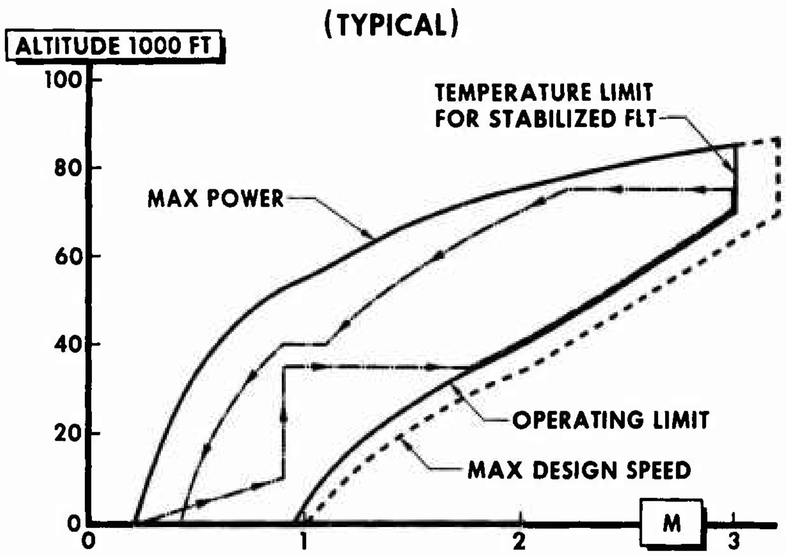


FIG. 5 SPEED-ALTITUDE ENVELOPE

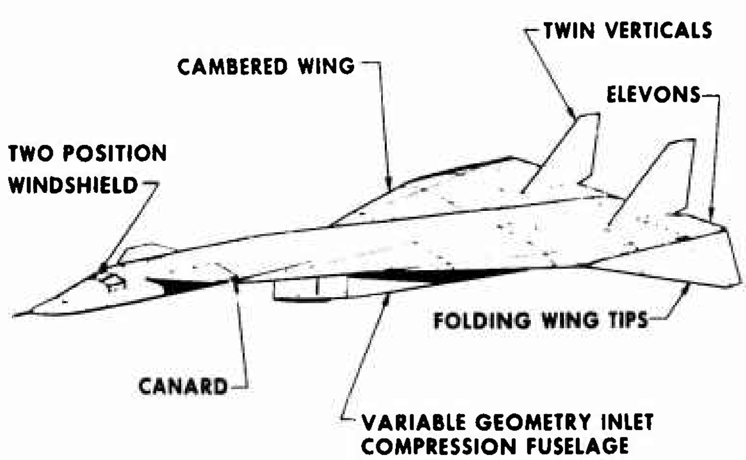


FIG. 6 AERODYNAMIC FEATURES

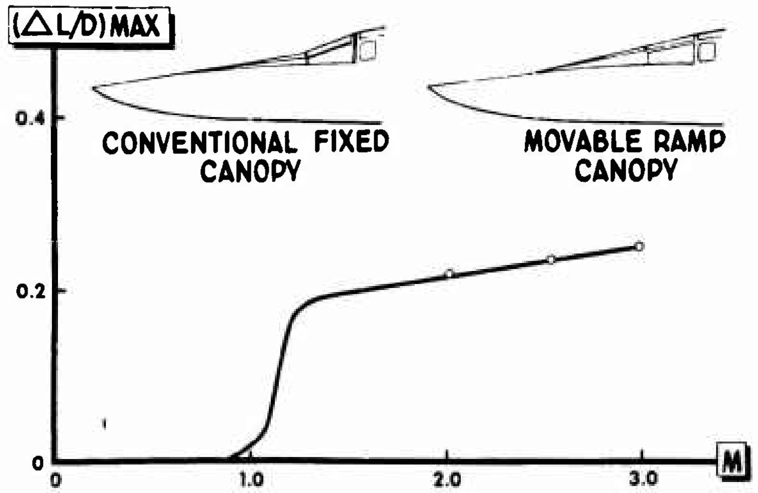


FIG. 7 CANOPY

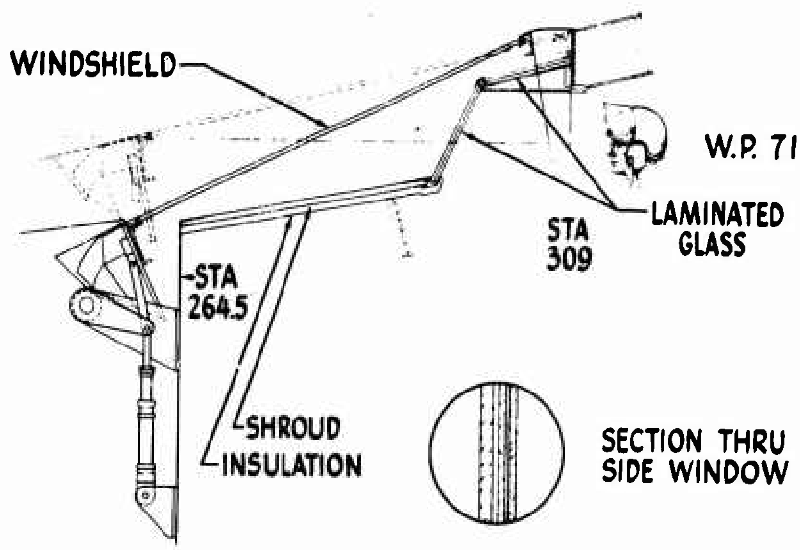


FIG. 8 CLEAR VISION DETAILS

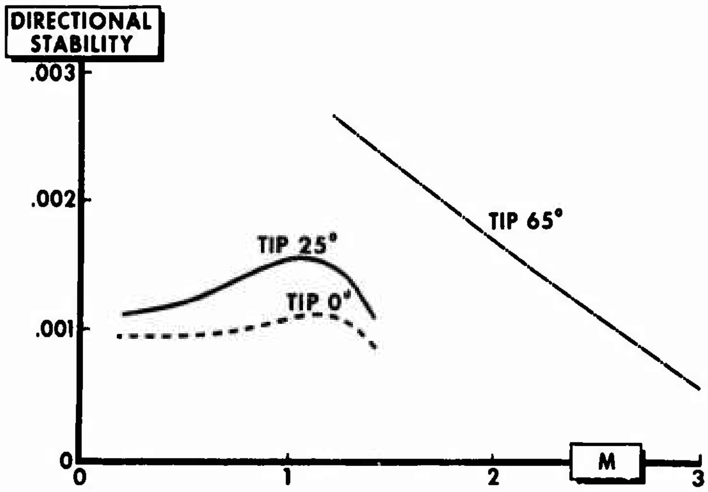


FIG. 9 DIRECTIONAL STABILITY

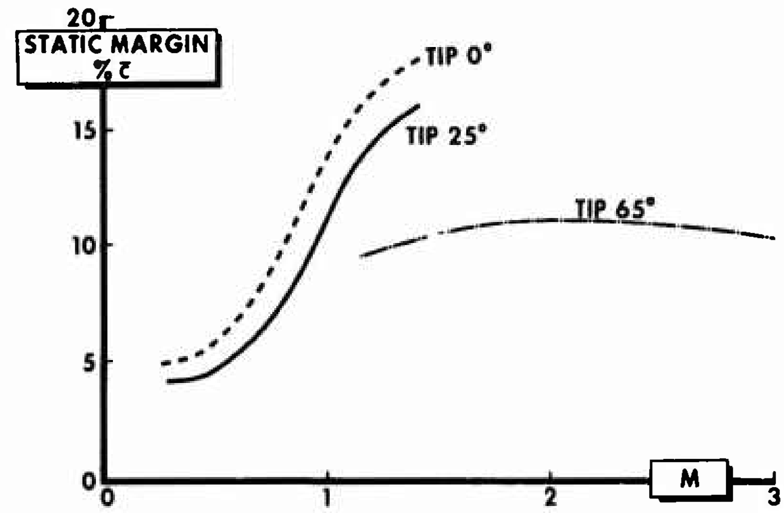


FIG. 10 LONGITUDINAL STABILITY

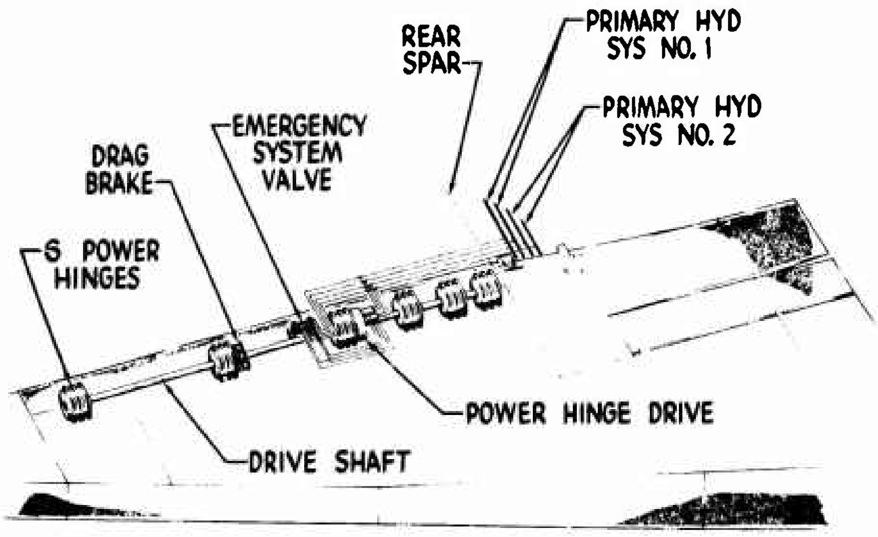


FIG. 11 WING TIP FOLD SCHEMATIC

- SIX ROTARY POWER HINGES IN-LINE PER WING
- COMMON SHAFTING DRIVEN BY DUAL HYDRAULIC MOTORS
- PILOT SELECTED WING FOLD POSITIONS
- 400°F OPERATING TEMPERATURE
- 2500:1 RATIO (TOTAL REDUCTION 32,000:1)
- H-11 STEEL CONSTRUCTION
- STRUCTURAL DEMONSTRATION TO 104% ULT LOAD
- FATIGUE TESTED TO 5000 HR OPERATIONAL CYCLE

FIG. 12 WING OUTER PANEL FOLD ACTUATION POWER HINGE

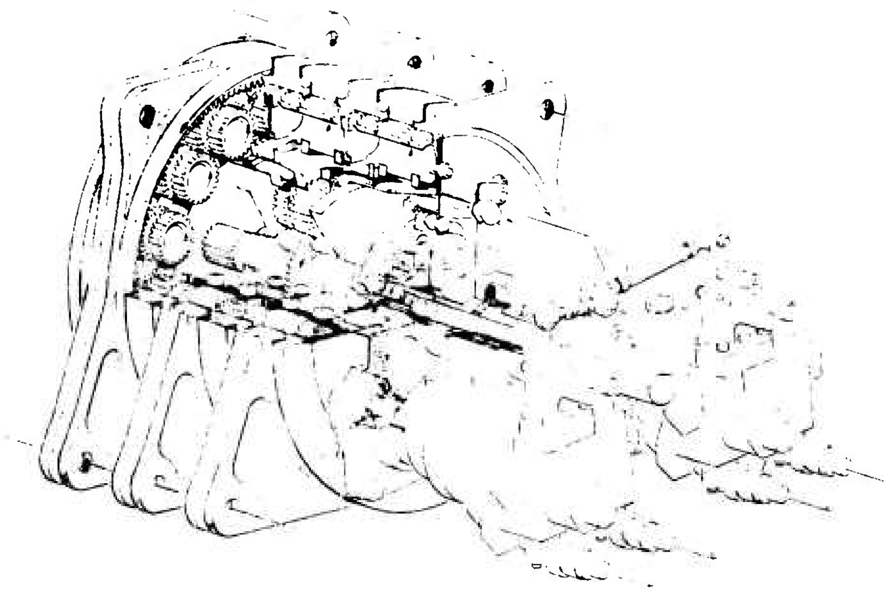


FIG. 13 POWER HINGE AND SPEED REDUCER CUTAWAY

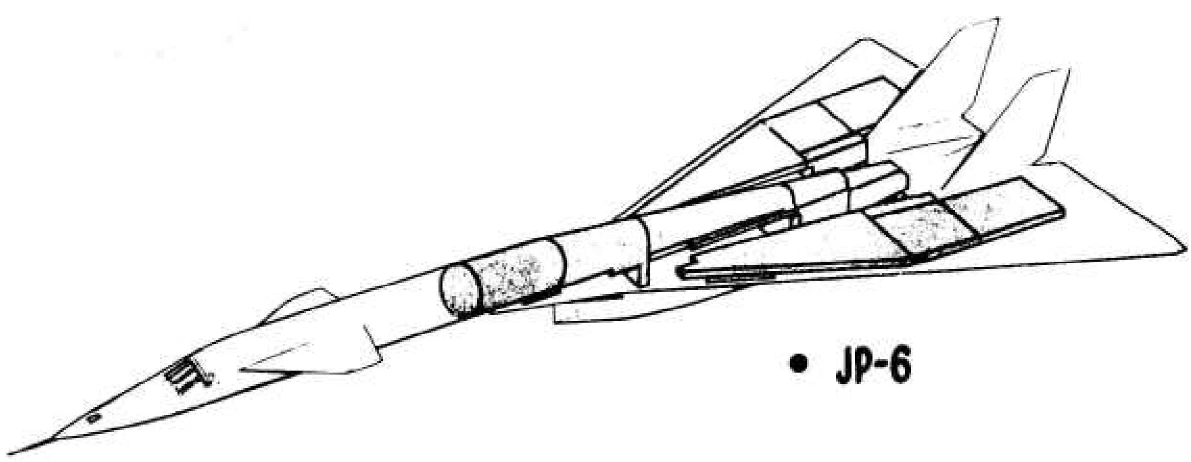


FIG. 14 AIR VEHICLE FUEL SYSTEMS

- SUPPLY JP-6 TO 6 ENGINES AT 300°F MAXIMUM
- SUPPLY JP-6 TO A/B AT 300° F MAXIMUM
- SIMPLICITY
- SEQUENCE FUEL TO MAINTAIN C.G. WITHIN LIMITS
- CONTROL TANK PRESSURES
- PREVENT AUTOGENOUS IGNITION
- GROUND PRESSURE REFUELING
- ICING PROTECTION
- PROVIDE HEAT SINK
- PROVIDE FILTRATION TO ENGINE

FIG. 15 FUEL SYSTEM REQUIREMENTS

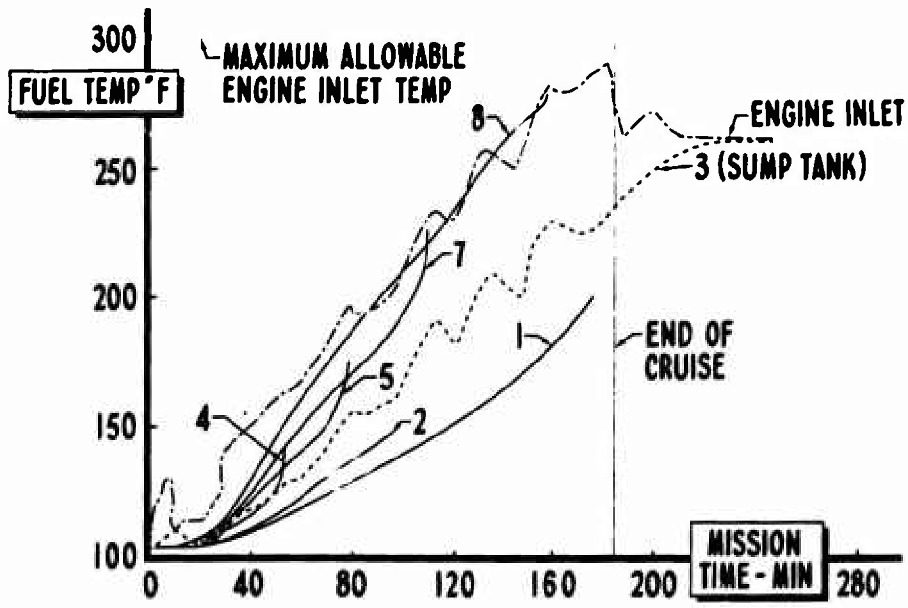
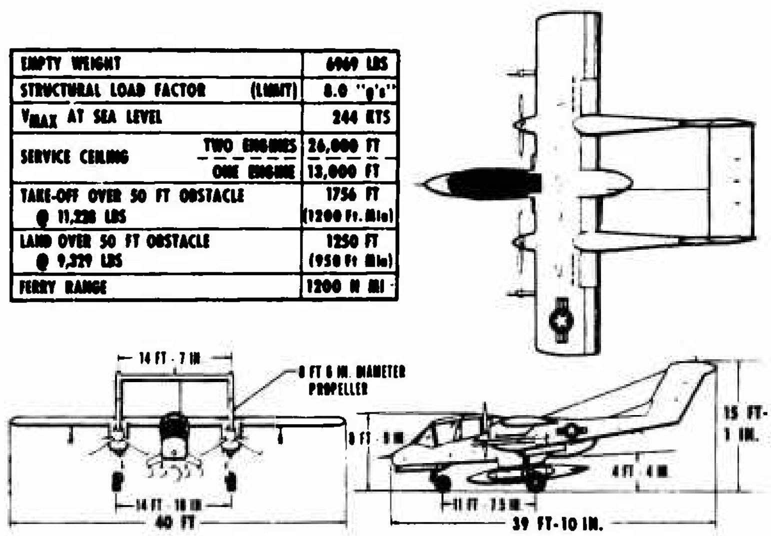


FIG. 16 **HEAT SINK JP-6**



EMPTY WEIGHT		6969 LBS
STRUCTURAL LOAD FACTOR	(LIMIT)	8.0 "g's"
V_{MAX} AT SEA LEVEL		244 KTS
SERVICE CEILING	TWO ENGINES	26,000 FT
	ONE ENGINE	13,000 FT
TAKE-OFF OVER 50 FT OBSTACLE @ 11,228 LBS		1756 FT (1200 Ft. Min)
LAND OVER 50 FT OBSTACLE @ 9,329 LBS		1250 FT (950 Ft Min)
FERRY RANGE		1200 N MI

FIG. 17 **THREE VIEW**

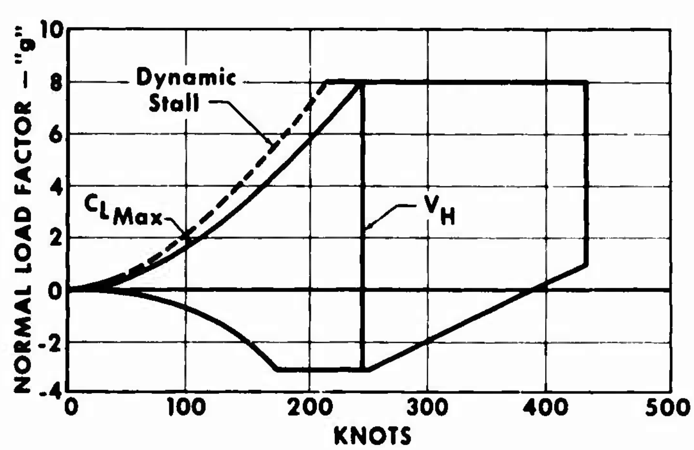


FIG. 18 **LOAD FACTOR DIAGRAM**

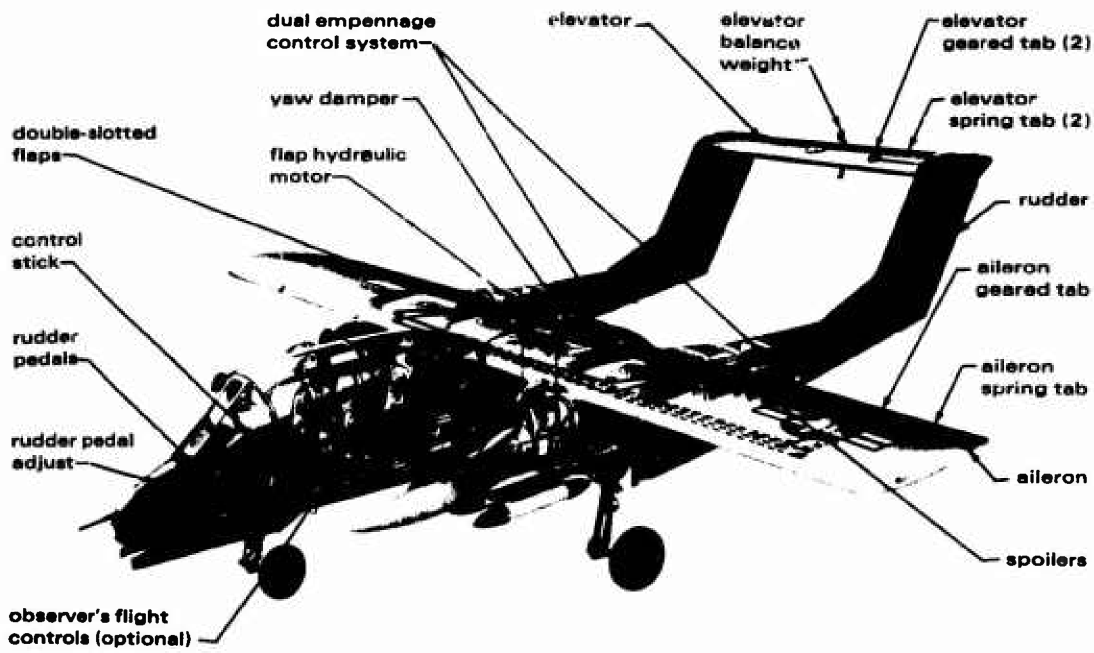


FIG. 19 FLIGHT CONTROLS

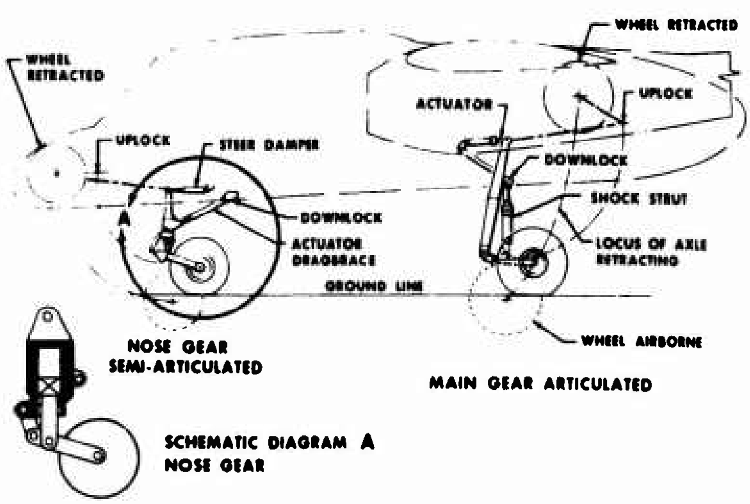


FIG. 20 LANDING GEAR GEOMETRY

ETUDES AVANCEES

DANS LE DOMAINE DES COMMANDES DE VOL

Communication de MM. CADO+ et BROIHANNE++

+ Ingénieur en Chef du SERVICE TECHNIQUE AERONAUTIQUE - FRANCE

++ Ingénieur en Chef à SUD-AVIATION - FRANCE

ETUDES AVANCEES DANS LE DOMAINE DES COMMANDES DE VOL

1e PARTIE

CONCEPTION DES COMMANDES DE VOL DES AVIONS DE COMBAT MODERNES

La recherche de performances de plus en plus élevées, dans des domaines de vol de plus en plus étendus pour les avions de combat modernes, a posé aux ingénieurs de l'aéronautique un certain nombre de problèmes liés à l'augmentation de la puissance nécessaire pour actionner les gouvernes et aux variations importantes des caractéristiques de stabilité et de maniabilité.

Après avoir indiqué les solutions partielles, envisagées séparément, de ces différents problèmes, des solutions globales seront présentées ainsi que l'évolution actuelle permise par les améliorations technologiques importantes dont l'électronique a bénéficié ces dernières années.

1. - PROBLEMES RENCONTRES SUR LES AVIONS MODERNES -

Parmi les problèmes rencontrés dans cette recherche de performances de plus en plus élevées, on peut citer, sans que cette liste soit exhaustive :

- l'augmentation de la puissance nécessaire pour actionner les gouvernes.
- des caractéristiques aérodynamiques variant considérablement d'un point du domaine de vol à un autre se traduisant :
  - par des stabilités dynamiques souvent marginales.
  - par des maniabilités très difficiles à homogénéiser par suite des variations importantes des efficacités de gouverne en fonction de la vitesse, de l'altitude et du nombre de mach.
- l'introduction de perturbations sévères, indépendamment des perturbations d'ordre aérodynamique dues à la turbulence de l'atmosphère, liées au nombre de mach, comme le recul du centre de poussée, ou liées aux changements de configuration de l'avion sur les appareils à géométrie variable.
- les déformations structurales.
- en corollaire les problèmes de sécurité, introduits par la mise en place de dispositifs plus ou moins sophistiqués, dont la fiabilité est loin d'être comparable à celle d'une simple timonerie mécanique.

2. - RAPPEL DE QUELQUES SOLUTIONS MISES AU POINT POUR RESOUDRE CES PROBLEMES -

Pour obtenir des qualités de vol convenables que des moyens purement aérodynamiques se montraient impuissants à obtenir, il a été fait appel : d'une part à des moyens essentiellement mécaniques et hydrauliques, d'autre part à des moyens électriques utilisant des chaînes de pilotage automatique : servo-amortisseurs et pilotage transparent.

C'est ainsi que pour vaincre les efforts importants nécessaires pour déplacer les gouvernes, on a été conduit à l'utilisation de servo-commandes hydrauliques, mais cette utilisation devait être complétée par la mise en place de systèmes de restitution d'efforts artificielle, donnant au pilote des efforts de manoeuvre qui ne peuvent plus lui fournir directement les forces aérodynamiques agissant sur les gouvernes ;

.../

C'est ainsi également que pour améliorer la stabilité de l'appareil, on a mis en place, en série dans les timoneries, des vérins commandés par des dispositifs servo-amortisseurs (de tangage, de roulis, de lacet).

2.1 - Commande de vol classique -

Ces différents développements ont conduit ces dernières années à ce que l'on pourrait appeler la commande de vol classique d'un avion moderne qui comprenait :

- le dispositif de commande à la disposition du pilote : manche ou volant et palonnier.
- des éléments de transmission des déplacements du manche aux gouvernes : câbles ou bielles.
- des organes de restitution d'efforts, utilisés conjointement avec des dispositifs de démultiplication variable ou de non linéarité pour tenir compte des variations d'efficacité des gouvernes et rétablir au niveau du pilote des efforts de manoeuvre homogènes dans tout le domaine de vol de l'avion.
- des vérins montés en série dans la timonerie, commandés par les dispositifs servo-amortisseurs.
- des organes de puissance constitués par des vérins hydrauliques asservis, actionnant directement les gouvernes.

Cet ensemble est présenté sur la figure (1). L'avantage principal de ces dispositifs à base mécanique réside dans leur simplicité de fonctionnement et de réalisation qui entraîne par voie de conséquence une bonne sécurité de fonctionnement.

L'inconvénient majeur provient du manque de souplesse de tels matériels dont les réglages sont fixes ou au mieux commandés en boucle ouverte à partir de certaines conditions de vol, et qui par suite ne sauraient être adaptés correctement pour tous les points d'un domaine de vol aussi étendu que celui d'un avion supersonique. Par ailleurs l'introduction de certains éléments dans la timonerie ne doit se faire qu'avec une certaine prudence car ces modifications risquent d'introduire des causes d'instabilité par couplage timonerie-avion (risques d'instabilité introduites par masse d'effort par g). Le bon fonctionnement d'une telle commande de vol implique enfin pour obtenir des caractéristiques de pilotage correctes, de limiter au maximum les frottements résiduels, les seuils, les hystérésis dans les timoneries mécaniques, et d'être aussi indépendant que possible des déformations structurales, problèmes qui ne sont pas toujours très faciles à résoudre.

Cependant une amélioration sensible a été apportée à l'utilisation de cette commande de vol classique, d'une part par l'utilisation en parallèle du pilotage transparent, dont les éléments sont venus se greffer sur les timoneries existantes, d'autre part par l'intégration des dispositifs servo-amortisseurs dans les vérins hydrauliques de gouvernes, premier pas vers la commande de vol électrique.

2.2 - Pilotage transparent -

Le pilotage transparent qui permet d'obtenir des qualités de pilotage en principe indépendantes du véhicule support, utilise une chaîne de pilotage automatique venant s'installer en parallèle sur la timonerie classique.

Ce principe de pilotage a été utilisé sur un certain nombre d'avions de combat français, sous le nom d'auto-commande de profondeur.

.../

2.2.1 - Fonctions de l'auto-commande de profondeur -

Les fonctions demandées à une telle chaîne de pilotage automatique sont les suivantes :

- obtenir dans tout le domaine de vol de l'appareil des efforts par g corrects et homogènes.
- maintenir l'assiette avec une stabilité convenable.
- réduire l'effet des couples perturbateurs.
- accessoirement, car très souvent réalisé par un servo-amortisseur de tangage distinct, amortir l'oscillation rapide d'incidence.

2.2.2 - Principes du pilotage transparent -

Le principe de pilotage consiste à détecter l'effort exercé par le pilote sur la timonerie et à commander un facteur de charge proportionnel à l'effort du pilote.

Pour cela deux procédés peuvent être utilisés :

- soit commander directement un facteur de charge mesuré par un accéléromètre normal,
- soit commander une vitesse angulaire modulée en fonction de la vitesse, cette vitesse angulaire étant mesurée par un gyromètre.

C'est cette deuxième méthode qui a été utilisée dans les réalisations actuellement en cours en France.

Compte tenu du retard de la variation du facteur de charge sur la vitesse angulaire, il est nécessaire d'introduire un terme d'avance de phase, surtout important en vol supersonique à haute altitude, pour obtenir, par l'intermédiaire d'une vitesse angulaire commandée fonction de l'effort, une variation du facteur de charge, proportionnelle à l'effort.

Le gyromètre est utilisé par ailleurs, après une double intégration, pour la stabilisation de l'assiette et la réduction de l'effet des couples perturbateurs en transonique.

2.2.3 - Réalisation -

La figure (2) donne le schéma général d'une auto-commande de profondeur, et l'implantation des différents éléments dans la timonerie.

On y distingue en particulier le dynamomètre, placé le plus près possible du pilote, qui mesure l'effort exercé par le pilote ainsi que le dispositif de trim automatique utilisé pour éviter tout à-coup au débrayage de l'auto-commande. Par ailleurs compte tenu des caractéristiques de pilotage particulièrement délicates des avions rapides, en particulier aux grandes efficacités de gouverne dans certaines conditions de vol, un certain nombre de dispositifs de sécurité ont été mis en place qui détectent les éléments ou les circuits défaillants et coupent la chaîne de pilotage automatique avant toute manoeuvre dangereuse de l'appareil.

Cette auto-commande est en effet utilisée dans tout le domaine de vol d'un appareil supersonique, du décollage à l'atterrissage, jusqu'à des facteurs de charge de l'ordre de 9.

2.3 - Intégration des servo-amortisseurs dans les vérins hydrauliques de gouverne -

Sur un appareil de combat supersonique, cette intégration a été réalisée par la mise en place d'une chaîne de pilotage électrique, doublée en cas de panne par la timonerie classique.

Les servo-commandes de gouvernes sont, dans ce but, électro-hydrauliques et reçoivent d'une part les ordres du pilote introduits par l'intermédiaire de potentiomètres détectant les variations de position des commandes de profondeur et de gauchissement, d'autre part les ordres provenant de servo-amortisseurs de roulis et de tangage. .../

L'avion étant un avion à aile delta, équipé d'élevons, ces ordres de profondeur et de gauchissement sont mélangés électriquement avant de venir attaquer les servo-commandes électro-hydrauliques d'élevons.

En cas de panne électrique, un système d'embrayage vient relier mécaniquement les servo-commandes de gouverne à la timonerie.

On a ainsi réalisé une véritable commande de vol électrique dont le secours est constitué par une timonerie mécanique.

La figure (3) donne le schéma d'ensemble de cette commande de vol. On y retrouve, en amont de servo-commandes intermédiaires, les différents éléments : système de restitution d'effort, trim électrique, de la commande classique.

Cette disposition, en éliminant les frottements de timonerie, et en intégrant les servo-amortisseurs dans les vérins de gouverne, a permis de réaliser un gain appréciable dans les qualités de vol de l'appareil (réponses plus rapides et plus précises) et dans le devis de masse de l'ensemble.

Ce sont par ailleurs ces principes de base qui ont été appliqués dans la conception des commandes de vol de l'avion de transport civil supersonique Concorde, décrites dans la deuxième partie de cet exposé.

3. - <u>CONCEPTION D'UN SYSTEME DE COMMANDE DE VOL MODERNE</u> -

Les études et réalisations précédentes dans le domaine des commandes de vol électriques et du pilotage transparent ont permis d'ores et déjà d'améliorer de façon sensible les qualités de vol des avions de combat.

Il reste cependant encore un pas important à franchir avant de parvenir à un système de commande de vol vraiment rationnel répondant à toutes les exigences entrainées par la recherche de performances de plus en plus élevées dans des domaines de vol de plus en plus étendus.

Un tel pas ne peut être franchi que par une intégration totale du pilotage transparent dans le système de commande de vol et la généralisation de la commande de vol électrique.

Le principe de base d'un tel système serait constitué par une commande directe, en boucle fermée, de la trajectoire de l'avion, dont les gouvernes reçoivent également par ailleurs tous les signaux nécessaires à l'amélioration de la stabilité de l'appareil, et à l'élimination des effets des perturbations extérieures.

Une telle conception implique de repenser complètement la disposition des organes de commande à la disposition du pilote, afin d'en rendre l'opération plus logique et plus naturelle. Elle implique également de mener en parallèle les études des sécurités nécessaires pour un fonctionnement aussi sûr que celui des commandes de vol actuelles.

Le schéma de la figure (4) indique quelle pourrait être la disposition d'un tel système de commande de vol.

La multiplicité des chaînes de commande, rendue nécessaire pour remplir les objectifs de sécurité ne peut bien entendu être réalisée de façon satisfaisante, que compte tenu de la miniaturisation des éléments calculateurs, rendue possible par les progrès actuels de l'électronique.

L'aboutissement d'un tel système constitue ainsi la synthèse des études menées aussi bien dans le domaine de l'aérodynamique et des qualités de vol que dans le domaine de la technologie avancée, hydraulique et électronique.

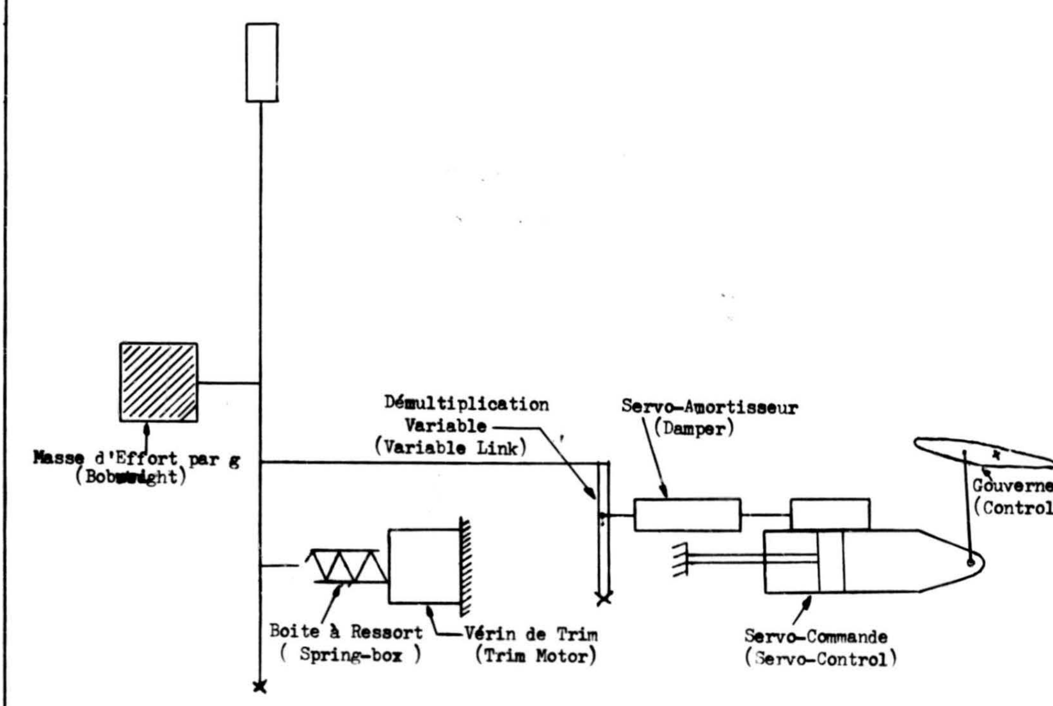
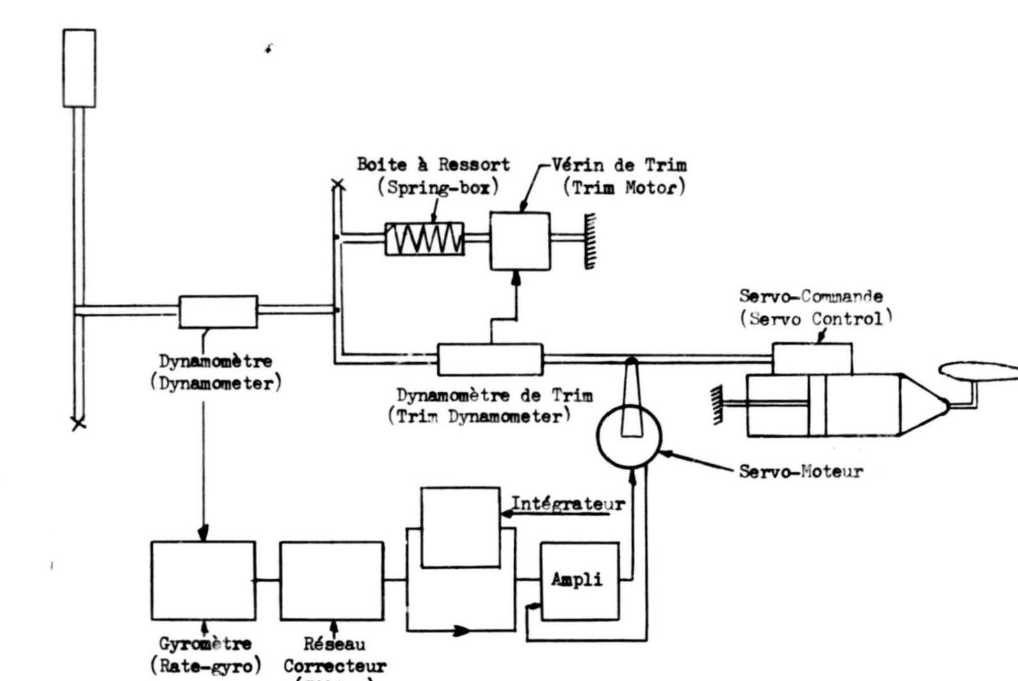


FIGURE 1.



Liaisons Electriques (Electrical Link)
Liaisons Mécaniques (Mechanical Link)

FIGURE 2.

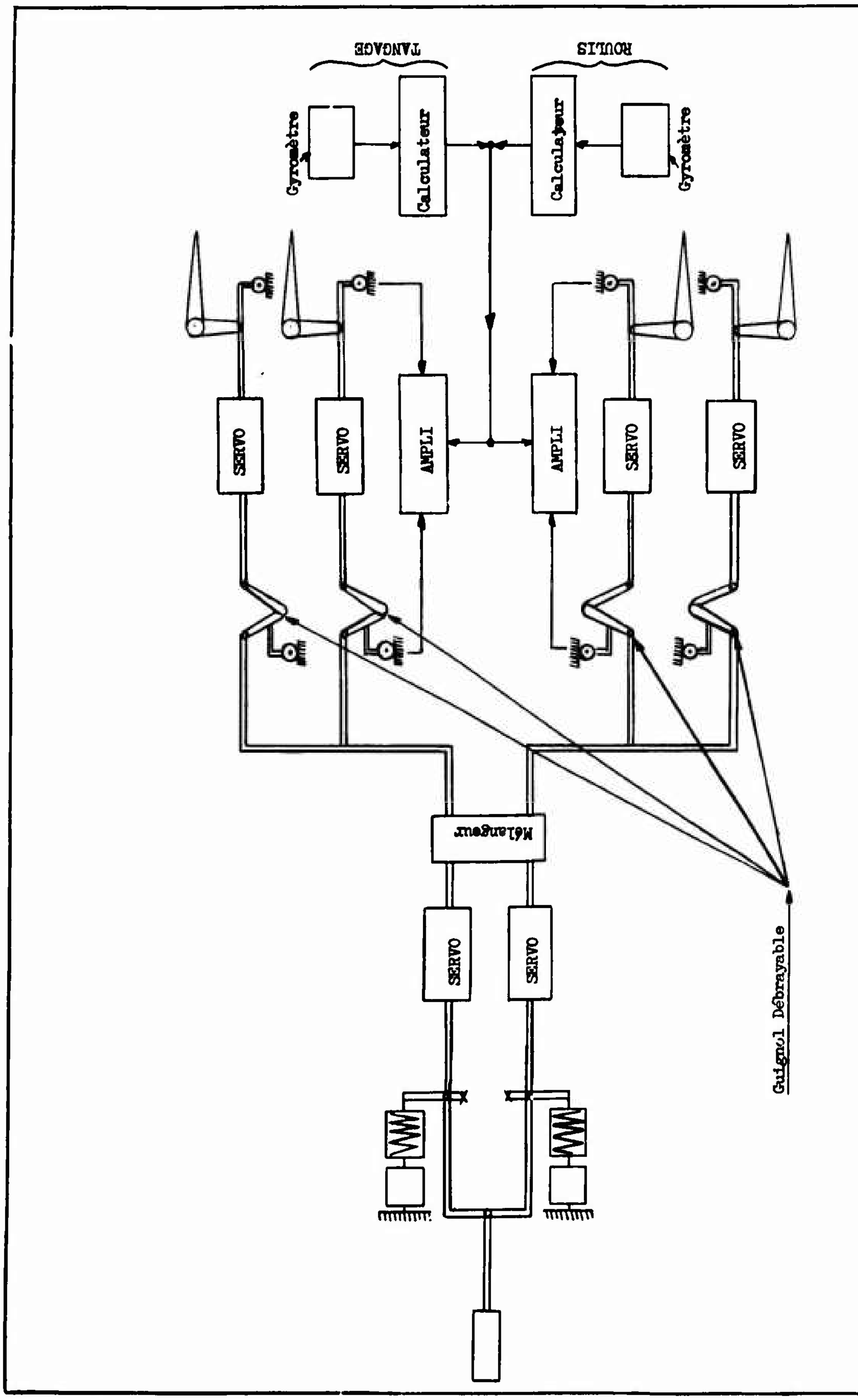

TANGAGE
ROULIS
Gyromètre
Calculateur
Calculateur
Gyromètre
SERVO
SERVO
AMPLI
AMPLI
SERVO
SERVO
Mélangeur
SERVO
SERVO
Guignol Débrayable

FIGURE III

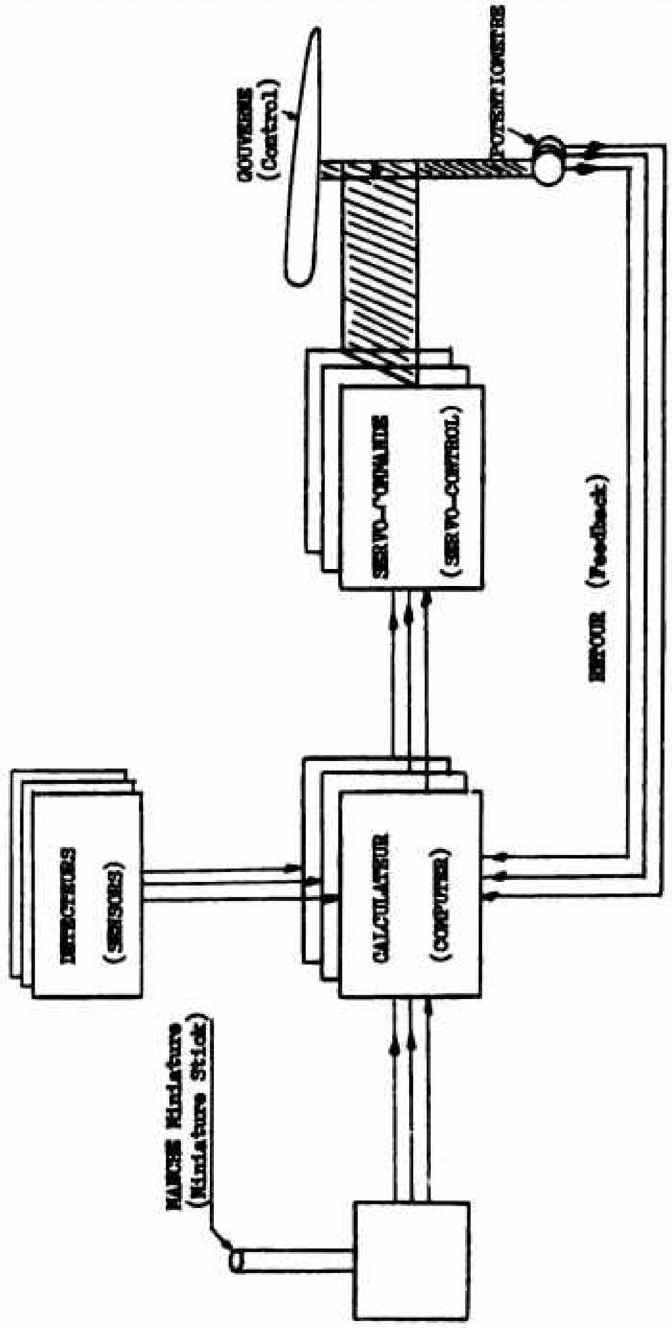

GOUVERNE
(Control)
POTENTIOMETRE
SERVO-COMMANDE
(SERVO-CONTROL)
RETOUR (Feedback)
DETECTEURS
(SENSORS)
CALCULATEUR
(COMPUTER)
MANCHE Miniature
(Miniature Stick)

FIGURE IV

ETUDES AVANCEES DANS LE DOMAINE DES COMMANDES DE VOL

2e PARTIE

LE SYSTEME DE COMMANDES DE VOL DE L'AVION DE TRANSPORT CIVIL SUPERSONIQUE CONCORDE

Les conceptions et les technologies avancées des avions militaires trouvent leur application, avec un certain décalage de temps pour des raisons d'économie industrielle, dans les avions civils. Mais ces conceptions doivent être revues à la lumière des règles de fiabilit' propres aux avions civils. De sorte que les conceptions civiles possèdent des aspects originaux que nous nous efforcerons de mettre en relief avec l'exemple du système de commandes de vol de l'avion de transport civil supersonique CONCORDE.

Les objectifs de performances et de fiabilité (au sens de la navigabilité et de l'utilisation opérationnelle) seront d'abord établis. Les conceptions de principe qui en découlent et certaines réalisations technologiques particulières seront ensuite présentées.

1. - OBJECTIFS DE PERFORMANCES -

Les objectifs de performances sont imposés par les qualités de vol et le pilotage, la résistance structurale et l'environnement.

1.1 - Qualités de vol -

Dans l'étendue du domaine de vol d'un avion supersonique, les lois de l'aérodynamique ont d'étranges conséquences. Aux basses vitesses : instabilité de vitesse entraînée par la variation inverse de la finesse avec la portance. Aux hautes vitesses subsoniques : hypersensibilité des gouvernes. Aux vitesses transsoniques : instabilité statique en vol équilibré provoquée par le déplacement du foyer. Aux vitesses supersoniques : limitation de manoeuvrabilité par la perte d'efficacité des gouvernes et l'accroissement des moments de charnières · perte de stabilité dynamique transversale, etc... A tous ces problèmes le système de commandes de vol doit faire face, pour réduire la tâche du pilote.

A la vérité les objectifs de sécurité et des considérations de nature psychologique ont conduit à imposer au TSS des qualités de vol acceptables en l'absence d'aides artificielles au pilotage. Mais il est apparu, en particulier par des études préliminaires sur un simulateur de vol, que les qualités de vol ainsi manifestées ne pouvaient être acceptées opérationnellement que dans des conditions occasionnelles, du fait de l'attention requise par le pilotage et de la charge de travail résultante, particulièrement dans les conditions atmosphériques turbulentes. Des dispositifs d'aides au pilotage sont donc nécessaires.

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1.1.1 - Qualités de vol de l'avion de base -

Les qualités de vol requises pour l'avion de base ne sont pas particulières au TSS, mais il est admis de les apprécier en relation avec les objectifs de sécurité (Voir chap. 2). L'avion doit donc pouvoir voler correctement et sûrement sans exiger une habileté exceptionnelle du pilote.

Portons seulement notre attention sur un problème spécifique concernant la manoeuvrabilité longitudinale dans le domaine transsonique et supersonique. Pour le TSS, le passage transsonique est marqué par un accroissement considérable de la marge statique, une perte d'efficacité des gouvernes et une augmentation de leurs moments de charnières (Fig. 1) telles que la manoeuvrabilité serait réduite à l'extrême sans un artifice, sauf au prix d'un dimensionnement exorbitant des gouvernes et de leurs vérins de manoeuvre. Cet artifice consiste à déplacer le centre de gravité au moyen d'un transfert de combustible de façon à conserver une marge statique normale. (Ceci a par ailleurs pour effet de réduire la traînée d'équilibrage en vol supersonique, et même de permettre une adaptation tridimensionnelle optimale des profils de voilure du point de vue de la traînée). Les objectifs de manoeuvrabilité fixés par les règlements officiels aux grandes vitesses imposent les facteurs de charge suivants, une certaine dérogation étant admise pour les phases de vol transsoniques (*) :

	Etats :	Fréquent	Occasionnel
Domaines de vol	Autorisé	0,5 - 1,6 (0,8 - 1,3) *	0,8 - 1,3 0,8 - 1,2) *
	Périphérique	0,8 - 1,3 (0,8 - 1,2) *	0,8 - 1,2

Les objectifs sont particulièrement critiques pour les conditions de vol occasionnelles, telles que résultant par exemple de la panne d'une source de puissance hydraulique. Ils ne peuvent être réalisés que grâce à un compromis entre la capacité de transfert de combustible (qui conditionne la disposition et le volume des réservoirs d'équilibrage, le débit des pompes de transfert), la puissance des vérins de commande des gouvernes (qui conditionne leur masse, la traînée de leur carénage, leurs performances dynamiques), et les centrages limites.

Bien entendu, la panne complète du transfert de combustible doit être considérée comme improbable dans le cas du transfert vers l'avant, ce qui exige une large redondance du système (triplication des pompes, largage de combustible en ultime secours). Mais en outre la décélération de l'avion doit être compatible avec le taux de transfert du point de vue des qualités de vol. Le ralentissement rapide sans transfert, nécessité par une urgence quelconque ou involontaire (panne totale de propulsion) doit permettre de conserver le contrôle de l'avion. Il doit donc exister un certain domaine subsonique permettant le vol à des centrages supersoniques pendant un intervalle de temps suffisant pour assurer le déplacement du centre de gravité. Ce domaine existe effectivement, d'après l'expérimentation effectuée sur un simulateur de vol. (Fig. 2).

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1.1.2 - Amélioration des qualités de vol avec les dispositifs artificiels -

Les objectifs de qualités de vol dans les conditions courantes, y compris les turbulences, impliquent l'utilisation d'aides au pilotage. Leur degré d'intégration avec le système de commandes doit être soigneusement étudié en fonction des objectifs de sécurité (Voir chap. 2).

1.1.2.1 - Automanette -

Une automanette est nécessaire pour rétablir la stabilité de vitesse, aux basses vitesses. En effet la divergence de vitesse en approche finale se traduit par le doublement d'un écart de vitesse en 20 secondes. Un amortissement correct, réduisant de moitié un écart de vitesse en 4 secondes, est obtenu avec une loi d'asservissement de la poussée de la forme (mode maintien de vitesse) :

$$T = \left[K_V \ \Delta V + K_o \ \theta \ \frac{Cs}{1 + Cs} \right] \ \frac{s + C_2}{s}$$

On peut noter que l'assiette longitudinale est utilisée pour corriger l'asservissement dans la mesure où cette action n'est pas défavorable de façon appréciable à la stabilité dynamique autour du centre de gravité.

1.1.2.2 - Autostabilisateurs (Fig. 3, 4) -

Des autostabilisateurs sont utilisés sur les 3 axes. Sur l'axe de tangage, l'objectif fixé initialement à basse vitesse était que 65 % du facteur de charge stabilisé soit obtenu en 2 secondes sur une action de la gouverne de profondeur. Cette condition avait été établie à priori, faute de critère règlementaire, pour parer au retard introduit par la déportance initiale des élevons et à la lenteur des mouvements autour du C.D.G due à la faible marge statique. Dans ce but une loi de précommande était envisagée. Mais il a été reconnu ultérieurement que cet objectif n'était pas fondamental et qu'il était préférable d'assurer un amortissement correct des mouvements de l'avion dans une turbulence, au moyen d'une loi simple du type :

$$q_{ST} = K_1 \ q + K_2 \ \frac{Cs}{1 + Cs} \ q$$

A grande vitesse l'objectif fixé est l'obtention d'un amortissement réduit de l'oscillation d'incidence supérieur ou égal à 0,6. La loi précédente y répond, moyennant une variation convenable des gains. D'autre part, l'autorité du stabilisateur a été fixée de façon à éviter la saturation dans une forte turbulence (probabilité 10^{-3}) pendant 95 % du temps.

Pour les axes transversaux, les objectifs fixés aux stabilisateurs correspondent à un amortissement réduit du roulis hollandais de 0,4 et à la réduction maximale de la susceptibilité de l'avion aux turbulences transversales, caractérisée par un taux d'inclinaison ϕ / V inférieur à 0,2. En outre il a été demandé aux stabilisateurs de minimiser les perturbations dynamiques résultant de pannes de moteurs. Mais il est apparu que l'action des stabilisateurs de roulis entraînait, dans leur limite d'autorité, une réduction excessive de maniabilité en gauchissement, qui devait être corrigée par une loi de précommande. Les lois de stabilisation retenues sont alors de la forme :

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$$p_{ST} = K_1\, p + K_2 \int p_c \qquad \text{en roulis}$$

$$r_{ST} = K \frac{Cs}{1 + Cs}\, r \qquad \text{en lacet,}$$

les gains et autorités variant avec le nombre de Mach.
Avec ces lois l'effet des pannes de moteur sur la réponse dynamique de l'avion est minimisée : l'action du stabilisateur de roulis pour limiter l'inclinaison latérale est remarquable. Cependant les stabilisateurs seuls ne se montrent pas suffisamment efficaces sous l'aspect particulier de la protection des entrées d'air contre l'effet d'un dérapage excessif lors d'une panne de moteur en vol supersonique. Il a fallu avoir recours à un dispositif spécial, braquant la gouverne de direction sous l'action directe d'un paramètre caractéristique d'une telle panne, c'est-à-dire en définitive la poussée, plutôt que sous l'action retardée du lacet ou du dérapage qui en résulte.

1.1.2.3 - Autotrim -

Malgré le déplacement du C.G., la marge statique ne peut être maintenue constante au passage transsonique, d'où il résulte une instabilité statique en vol équilibré (Fig. 5).

Cette instabilité fait l'objet de controverses. La règlementation américaine préconise qu'il y soit remédié. Dans ce but un trim automatique asservi au nombre de Mach modifie l'équilibre de la commande de profondeur de façon à exiger du pilote une poussée sur le manche croissant conventionnellement avec la vitesse. Mais certains pilotes estiment ce dispositif sans intérêt, du fait du caractère temporaire du passage transsonique, de la lenteur de la divergence, de la gêne créée par la nécessité d'ajuster en permanence l'équilibre de la commande à effort nul.

1.2 - Pilotage -

Considérons deux aspects typiques des objectifs de performances du système de commandes de vol imposés par le pilotage : la précision et l'autoadaptation des lois d'efforts artificiels.

1.2.1 - Précision -

Le contrôle des braquages de gouvernes doit être précis en pilotage manuel dans les phases de vol subsoniques de grande sensibilité de gouverne : vols à basse vitesse avec une marge statique minimale, vols à grande vitesse avec une efficacité de gouverne maximale. Pour qu'il en soit ainsi l'erreur de commande attribuable aux éléments situés dans la chaîne de commande entre les vérins de puissance et le dispositif d'effort artificiel et de trim doit être minimale. Avec une commande électrique cette règle de l'art est satisfaite d'elle-même car la précision d'une chaîne de synchrodétection est excellente et indépendante de la longueur de la chaîne.

Les systèmes automatiques, d'autre part, ne s'accommodent pas d'une commande imprécise, contrairement à l'opinion courante. La simplicité, l'efficacité et la stabilité du pilote automatique et des autostabilisateurs sont liées à une faible valeur de l'hystéré-

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sis des organes de puissance. C'est pourquoi les ordres de stabilisation sont injectés directement au niveau des servovalves des vérins de commande de puissance, dont les performances statiques et dynamiques sont excellentes (Voir Chap. 3.3).

Les ordres du pilote automatique ne sont pas injectés au même niveau, mais au niveau de vérins relais, pour réaliser l'objectif de sécurité d'une limitation d'autorité mécanique. Les hautes performances statiques et dynamiques de ces vérins relais et des chaînes de commande électrique confèrent à l'ensemble de la chaîne de pilotage automatique les performances de précision exigées. Les problèmes de stabilité qui apparaissent normalement avec des systèmes de ce niveau de performance statique sont résolus par l'adjonction de filtres hydrauliques, pour les vérins de puissance, et par un choix judicieux des gains de ces vérins. Aucune oscillation limite mesurable n'apparaît.

1.2.2 - Autoadaptation des efforts artificiels -

Les dispositifs d'efforts artificiels d'une commande classique, telle qu'elle a été choisie pour le TSS, réalisent une liaison cinématique déterminée entre le pilote et les gouvernes. Ils ont essentiellement pour rôle de donner au pilote les informations musculaires immédiates pour le dosage de son action, les informations à plus long terme étant fournies par les instruments de pilotage et les sensations musculaires et visuelles résultant de la réponse de l'avion. Ces informations immédiates doivent être programmées en fonction des conditions de vol.

Or celles-ci sont nombreuses et complexes (centrage, masse, vitesse, nombre de mach, altitude...) et il n'est pas raisonnable d'envisager le calcul des informations nécessaires par un programme complexe impliquant des capteurs et des calculateurs appropriés. Il est nécessaire de déterminer des paramètres d'influence synthétiques.

Pour assurer le contrôle en tangage, l'information la plus utile est le taux de variation d'assiette ou le facteur de charge. L'objectif fixé est donc un taux d'effort par unité d'accroissement du facteur de charge déterminé dans tout le domaine de vol. Le problème consiste à programmer une relation simple entre cette information et l'effort artificiel. La solution retenue est fondée sur une remarque élémentaire : le braquage est proportionnel au facteur de charge, le coefficient de proportionnalité étant le braquage d'équilibre pour n = 1. Il en serait ainsi pour un avion idéal possédant une symétrie aérodynamique en tangage, ce qui est approximativement le cas d'un avion dans la configuration de croisière.

Ainsi donc pour réaliser l'objectif $\frac{\Delta F}{\Delta n} = Cte$

il suffirait, compte tenu de la remarque $\Delta \delta q = \delta q_{n=1} \quad \Delta n$

de réaliser une loi d'effort $\Delta F = \frac{Cte}{\delta q_{n=1}} \quad \Delta \delta q,$

le coefficient de proportionnalité variant comme l'inverse du braquage réalisant l'équilibre du tangage pour n = 1. Pratiquement on peut utiliser la valeur du braquage de trim d'effort δq_T comme mesure du braquage d'équilibre. D'autre part on tient compte

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de la non symétrie aérodynamique par une correction δq_o :

$$\Delta F = \frac{Cte}{\delta q_T - \delta q_o} \Delta \delta q$$

Enfin l'objectif de sécurité imposant une certaine contribution purement mécanique, la loi d'effort prend la forme suivante, séparant les contributions mécanique et électro-hydraulique

$$\Delta F = k_1 \Delta \delta q + \left[\frac{k_2}{\delta q_T - \delta q_o} - k_1 \right] \Delta \delta q$$

Contributions : mécanique + électrohydraulique

La réalisation pratique de cette loi fournit pour le TSS des résultats conformes à l'objectif. Alors que les taux de braquage par rapport au facteur de charge varient de 1 à 4 dans le domaine de vol, le taux d'effort reste voisin de l'optimum fixé de 30 daN/g (Fig. 6). Le dispositif réalise une autoadaptation remarquable aux conditions de vol les plus diverses.

1.3 - Résistance structurale -

La résistance structurale du TSS impose au système de commandes de vol des contraintes de définition. Tout d'abord au contraire d'un avion militaire de mêmes performances, la manoeuvrabilité doit être limitée au strict objectif des qualités de vol, sous peine d'une pénalité inutile pour la structure et les sources de puissance. Tout système de limitation à butée mécanique étant proscrit par sécurité, la limitation de manoeuvrabilité a été recherchée par les efforts artificiels. D'autre part la conception aérodynamique et structurale (voilure mince, fuselage effilé) imposée par les performances, amplifie les phénomènes aéroélastiques générateurs de couplages dynamiques pour les dispositifs automatiques.

A cause de sa formule géométrique et aérodynamique, la résistance du TSS est conditionnée en tangage par les manoeuvres, avec un facteur de charge limite 2,5, plutôt que par les rafales. Or la manoeuvrabilité, déterminée par les conditions aérodynamiques trans et supersoniques est surabondante en régime subsonique, spécialement à basse vitesse, et doit être limitée. Parmi les solutions possibles et compte tenu du choix d'une liaison cinématique directe et invariable entre le pilote et les gouvernes, il a été choisi d'assurer cette limitation au moyen du dispositif d'effort artificiel. La solution retenue assurant l'autoadaptation du taux d'effort relatif au facteur de charge résoud simplement le problème, en donnant au pilote une indication musculaire et invariable du facteur de charge de manoeuvre dans tout le domaine de vol.

En roulis la limitation de manoeuvrabilité assurée par un dispositif d'effort artificiel classique est satisfaisante.

En lacet une limitation programmée du braquage du gouvernail de direction est nécessaire. A une butée mécanique variable, a été préférée, par sécurité, un saut d'effort de 10 daN dont la perception suffit au pilote pour lui donner l'indication de la limite.

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Quant aux effets aéroélastiques, ils ont fait l'objet d'exposés au symposium d'Avril 1969 de l'AGARD à Marseille. Notons seulement que l'existence de modes de vibration d'assez basses fréquences ne doit pas être négligée dans la définition des dispositifs automatiques et spécialement les autostabilisateurs. Les couplages qu'ils peuvent provoquer sont à éliminer grâce à une connaissance précise des formes des modes structuraux, par un choix judicieux de l'implantation des gyromètres et des plate-formes gyroscopiques, et par l'incorporation de filtres appropriés.

1.4 - Environnement -

L'environnement du TSS est nettement plus sévère que celui des avions subsoniques actuels.

L'échauffement cinétique est appréciable, bien que la température maximale opérationnelle soit limitée à 400°K. Les servocommandes de puissance, par exemple, avec leurs synchrodétecteurs et leurs servovalves, doivent pouvoir supporter cette ambiance pendant 20.000 heures. A l'opposé des températures de - 40° peuvent être rencontrées pendant les phases de vol subsoniques. Dans cette gamme de températures, les réglages et la précision doivent rester sensiblement constants. Les matériaux des joints doivent conserver leur qualité et le fluide hydraulique doit conserver ses caractéristiques.

De même les hautes vitesses (V_{MO} = 530 kts C.A.S en début de croisière) et le haut niveau de propulsion induisent une ambiance vibratoire acoustique et structurale qui conditionne la définition structurale des gouvernes et de leur commande.

2. - OBJECTIFS DE FIABILITE -

Le système des commandes de vol est conçu pour répondre aux objectifs de fiabilité imposés par la sécurité et l'utilisation opérationnelle d'un avion de transport civil moderne.

2.1 - Objectifs et principes de sécurité -

Les règlements de navigabilité applicables au TSS, établis par l'AIR REGISTRATION BOARD britannique et le SERVICE TECHNIQUE AERONAUTIQUE français, considèrent la sécurité sous un jour nouveau. Le système des commandes de vol est en effet considéré sous l'aspect synthétique de sa contribution à la navigabilité et spécialement aux qualités de vol de l'avion, compte-tenu des conditions de pilotage. La sécurité de l'avion est d'autre part analysée d'un point de vue probabiliste, tempéré par le bon sens.

Ainsi :

a) de bonnes qualités de vol sont requises dans les conditions courantes. Ceci signifie que le comportement de l'avion doit correspondre au 1er, 2e et 3e degrés de l'échelle de COOPER dans toutes les conditions (domaine de vol et état de l'avion et des systèmes, conditions atmosphériques) dont la probabilité d'occurence est supérieure à $10^{-3,5}$.

b) une dégradation progressive des qualités de vol est admise dans des conditions occasionnelles. Ceci correspond à des degrés de l'échelle de COOPER compris entre le

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4e et le 6e pour des probabilités comprises entre $10^{-3,5}$ et 10^{-7}.

c) un taux global de pannes catastrophiques dues à la navigabilité extrêmement rare, c'est-à-dire inférieure à 10^{-7} par heure de vol, est exigé

d) le risque moyen au cours d'un atterrissage automatique doit avoir une probabilité inférieure à 10^{-7}, et le risque au cours d'un atterrissage particulier, lorsque les conditions sont connues, ne doit pas avoir une probabilité supérieure à 3.10^{-6}

A ces règles s'ajoutent des considérations de bon sens tenant compte de l'incertitude des données ou des estimations probabilistes, au stade d'un projet ou au début de l'exploitation d'un avion, ou simplement de l'impossibilité pratique de démontrer les taux d'occurence les plus faibles des conditions exceptionnelles. Ainsi :

a) Aucune panne simple ou combinaison de pannes non improbable ne doivent avoir un effet catastrophique

b) Des pannes rares ne doivent pas avoir d'effets dangereux

c) Des pannes courantes ne doivent avoir que des effets mineurs

En outre les possibilités d'erreurs de pilotage, de contrôle ou d'entretien, ne doivent pas être négligées.

2.2 - Applications des règles de sécurité -

Indiquons sommairement certaines des principales conséquences de ces règles sur la conception du système des commandes de vol.

2.2.1 - Justification d'une commande électrique -

Le principe d'une commande électrique ayant été choisi par des considérations de performances, il est fondamental de le justifier du point de vue de la navigabilité, à cause de sa nouveauté. On admet comme objectif spécifique des commandes de vol un taux de pannes catastrophiques de 10^{-9} pour satisfaire l'objectif d'un taux global de 10^{-7} pour l'avion.
Or l'expérience d'avions militaires, pour aussi notable qu'elle soit, ne peut fournir une assurance suffisante pour considérer les objectifs fixés, et spécialement le taux de panne catastrophique, comme techniquement réalisables aujourd'hui au moyen d'une commande purement électrique. En effet en admettant même qu'ils le soient en principe au prix d'un certain degré de redondance, il apparait comme impossible d'être assuré que l'analyse de sécurité d'un système électrique ainsi redondant, aussi minutieuse soit-elle, ne néglige aucune panne subtile de conséquence catastrophique. En tout cas on ne peut prétendre convaincre les Autorités Officielles qu'il puisse en être ainsi. Dans ces conditions, une solution de compromis a été choisie, comprenant une commande électrique doublée et surveillée et une commande mécanique de secours avec commutations automatiques. La probabilité de perte totale de la commande électrique est estimée pour les élevons à 2.10^{-8} par heure, pour des vols de 3 heures, et à 8.10^{-8} pour la direction. La panne catastrophique totale de l'ensemble des commandes électrique et mécanique est alors improbable.

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En outre, la fréquence de la panne de la commande électrique, qui entraîne une détérioration importante mais non catastrophique des qualités de vol par suite de la perte corrélative des autostabilisateurs, est acceptable au sens des règlements.

Ces valeurs sont obtenues d'une part au moyen d'une redondance (2 chaînes indépendantes, y compris leurs alimentations électriques, comportant chacune 3 chaînes élémentaires pour les 3 paires d'élevons et 2 chaînes élémentaires pour les 2 tronçons de gouvernail de direction) et d'autre part grâce à la simplicité des chaînes et à la fiabilité de leurs composants. Le taux de panne d'une chaîne élémentaire de commande d'une paire d'élevons comprenant 3 resolvers, un amplificateur et une servovalve, est de l'ordre de 10^{-4}. Une attention particulière est portée aux convertisseurs statiques assurant l'alimentation autonome des chaînes : chaque convertisseur et son boîtier de protection ont un taux de panne estimé de 8.10^{-5} par heure. Une démonstration de fiabilité exécutée sur 5 convertisseurs pendant 2500 heures sans panne permet déjà de garantir un MTBF de 12.500 heures avec un coefficient de confiance de 65 %.

Les pannes partielles de la commande électrique n'ont pas de conséquences ou ont des conséquences admissibles relativement à leur fréquence sur les qualités de vol et le pilotage. Ainsi la perte de l'une des 2 chaînes électriques n'a pas de conséquence, grâce aux commutations automatiques.

Le système de surveillance et de commutation des chaînes de commande doit être traité avec le même souci de fiabilité. La non détection d'une panne entraînant un embarquement de gouvernes, présente un risque majeur. La surveillance doit donc avoir un taux de panne extrêmement rare et être par suite doublée. Il en est de même du système de commutation, et spécialement des électrovannes alimentant les servovalves. De plus la commutation doit être automatique et extrêmement rapide pour limiter les embarquements de gouvernes à des valeurs admissibles pour la résistance structurale. Enfin les commutations ne doivent pas être intempestives : la surveillance ne doit agir qu'au delà d'un seuil d'écart compatible avec les tolérances.

D'autre part la commande électrique du TSS reste conventionnelle par le fait qu'elle réalise une liaison cinématique directe entre les ordres du pilote et les actionneurs des gouvernes. Il n'a pas été tiré un avantage technologique de cette liaison électrique pour réaliser un filtrage direct des ordres de pilotage, en vue d'optimiser la réponse de l'avion. Un tel filtrage, dont l'autocommande est un exemple d'application impliquerait une pleine autorité du filtre. Les problèmes de sécurité qui en résulteraient et leurs solutions certifiables sont apparus au premier examen comme assez complexes, pour ne pas permettre une application immédiate.

2.2.2 - Conception des aides au pilotage -

La panne d'un dispositif d'aide au pilotage ayant une conséquence notable sur le pilotage et les qualités de vol doit être occasionnelle. Cet objectif s'applique aux autostabilisateurs (sur un ou plusieurs axes), au dispositif de sensation musculaire (sur un ou plusieurs axes), au trim automatique. Il entraîne pratiquement la duplication de ces éléments.

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Les centrales aérodynamiques qui règlent les variations des gains de ces dispositifs sont elles-mêmes doublées et partiellement autosurveillées. Mais comme il n'est pas possible d'attendre de ces centrales une fiabilité telle qu'elle n'affecte pas notablement celle de la stabilisation les stabilisateurs sont conçus pour survivre à une panne détectée de centrale aérodynamique, mais avec des autorités et des gains réduits prédéterminés.

D'autre part, il est important de noter que l'action des stabilisateurs s'exerce au niveau des chaînes électriques des commandes de vol. La fiabilité de ces chaînes est en effet compatible avec l'objectif fixé et ne modifie pas notablement la fiabilité globale. De ce fait la réalisation technologique des stabilisateurs est très simple.

Quant au système d'effort artificiel, la perte totale, considérée comme présentant un risque majeur, ne peut être rendu improbable avec un système électro-hydraulique doublé. Il est donc imposé qu'une partie de l'effort soit réalisée par un élément mécanique (bielle à ressort).

Mais de même qu'une commande purement électrique n'a pas été retenue pour des raisons psychologiques plutôt qu'objectives, de même il a été considéré comme fondamental que l'avion reste pilotable en l'absence d'une ou plusieurs de ces aides au pilotage, quelle que soit la rareté de leurs pannes. Il est cependant admis que la charge de travail puisse être notablement accrue pour l'équipage, en particulier en présence de très fortes turbulences. Sous cet aspect le TSS reste un avion conventionnel.

Mais il ne suffit pas que l'avion soit réputé pilotable sans aides spéciales. Il faut aussi définir ces aides de telle sorte que leur panne n'ait pas de conséquences directes incontrôlables. C'est pourquoi ces aides doublées sont en outre autosurveillées avec une commutation et une déconnexion automatique, la probabilité des non-commutations par défaut de surveillance étant extrêmement rare ou improbable. D'autre part, ces aides ont une limitation d'autorité. Cette limitation est déterminée par un compromis entre les qualités de vol et la sécurité vis à vis de pannes actives. Pour les stabilisateurs les braquages de gouvernes commandés sont limités à 1 à 4° suivant l'axe et le nombre de Mach. Pour l'autotrim, la limitation porte sur la vitesse de variation de la correction de braquage.

2.2.3 - Conception du pilote automatique et du système d'atterrissage par mauvaise visibilité -

La liaison du pilote automatique et du système de commandes de vol est définie en fonction des objectifs de sécurité imposés. Il a été considéré que la sécurité vis à vis d'une panne catastrophique ne pouvait être garantie essentiellement par un dispositif automatique de surveillance purement électronique assurant la déconnexion du pilote automatique avec la fiabilité requise sauf pendant un court instant au cours de la phase d'atterrissage automatique, sous réserve d'un test préalable. Il a donc été décidé d'assurer la sécurité pendant la majeure partie du vol par un dispositif mécanique de limitation d'autorité.

L'étude de la sécurité pendant l'atterrissage automatique nécessiterait un long développement, car le système d'atterrissage automatique intègre de nombreux éléments, dont le système de commandes de vol et le pilote automatique ne sont que des éléments parmi les plus importants. Le sujet a été exposé au symposium de l'AGARD des 20 au 23 mai 1969

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à Cambridge. Notons seulement que les exigences de sécurité mentionnées précédemment sont considérées comme pouvant être satisfaites avec un ensemble complètement doublé, avec commutations automatiques, comportant notamment un pilote automatique autosurveillé agissant sur le système électrique de commande de vols surveillé.

Considérons en effet l'objectif d'un taux d'accident moyen de 10^{-7} par atterrissage, essentiellement dû au système d'atterrissage automatique. Attribuons aux pannes de matériel une contribution maximale de 50 %, le reste étant dû à des écarts excessifs de performances. Parmi les pannes de matériel admettons que la moitié au plus soit due aux équipements de bord. Ces pannes peuvent consister en la perte du guidage par la panne du système doublé (due à la panne d'un élément de chaque chaîne) ou en une panne active due à la panne non détectée de la chaîne travaillante. Admettons l'équi-partition de ces taux de panne. Considérons le premier cas dont la probabilité est $0,125.10^{-7}$ et calculons simplement la fiabilité d'une chaîne satisfaisant à l'objectif en considérant que la durée du risque est de l'ordre de 12 secondes par atterrissage. Le MTBF de chacune des 2 chaînes doit être au moins tel que

$$\left(\frac{12}{3600} \quad \frac{1}{MTBF} \right)^2 = 0,125.10^{-7}$$

soit environ 30 heures. Cette fiabilité est certainement réalisable et démontrable. En utilisant des éléments ayant des fiabilités élémentaires courantes, une fiabilité globale très supérieure peut être espérée.

2.3 - Objectifs opérationnels -

Les objectifs de fiabilité opérationnelle sont généralement contradictoires avec ceux de la navigabilité. Car les taux de sécurité extrêmes exigés par la navigabilité impliquent une certaine complexité des redondances et de surveillances nécessaires, alors que l'utilisation opérationnelle optimale implique le maximum de simplicité. Mais cette contradiction ne doit pas être trop schématisée. En effet, bien que la définition de principe d'un système soit liée aux objectifs de performances et de sécurité, les objectifs opérationnels peuvent être réalisés au moyen d'une technologie appropriée des éléments du système. Ces objectifs concernent en effet le contrôle du matériel à bord et son entretien en atelier. Il n'est pas question de pouvoir contrôler dans un délai acceptable l'intégrité de systèmes complexes autosurveillés, comportant en particulier des comparateurs de chaînes actives et de chaînes de surveillance, au moyen de manoeuvres classiques de check list. Le contrôle doit être rapide, ne doit nécessiter aucun matériel annexe et doit fournir une indication suffisante pour localiser un élément en panne. Dans ce but, le système de commandes de vol et des aides au pilotage est conçu pour être testé automatiquement, soit au moyen d'un système central à programme séquentiel soit au moyen de circuits d'autotest particuliers à chaque sous-système.

3. - CONCEPTIONS DE PRINCIPE ET TECHNOLOGIES -

Indiquons maintenant comment les objectifs de performances et de fiabilité énoncés sont appliqués à la conception générale du système de commandes de vol et à la définition .../

technologique de certains éléments importants.

3.1 - Schéma général du système de commandes de vol -

Les gouvernes comprennent les élevons assurant le tangage et le gauchissement, constitués de 3 tronçons par côté de voilure, et le gouvernail de direction, constitué de 2 tronçons. Chaque tronçon est actionné par une servocommande électro-hydraulique à double corps en tandem.
Les élevons se divisent en deux groupes contrôlés indépendamment internes d'une part, médians et externes d'autre part. Les premiers ont pour fonction principale le contrôle en tangage, les autres en gauchissement (Fig. 7).

Les commandes comportent des organes de pilotage classiques (manche, volant, pédales). Mais l'action du pilote sur les vérins de puissance s'exerce suivant deux modes. Le mode électrique est le mode normal. Il est doublé et surveillé. Le mode mécanique est utilisé en secours. Une séquence automatique assure, en cas de pannes, les commutations du premier mode électrique sur le second puis sur le mode mécanique. L'action de l'autopilote s'exerce par l'intermédiaire du mode électrique (Fig. 8).

3.1.1 - Dans le mode normal de commande manuelle électrique, les actions du pilote sont transmises directement à des chaînes doublées de synchrodétection. Ces chaînes n'introduisant que des erreurs minimes de positionnement, la précision de positionnement des gouvernes dépend de celle des organes de pilotage. C'est pourquoi le dispositif de sensation artificielle et le trim associé sont placés à l'amont de la commande.
Pour les élevons, les chaînes réalisent électriquement les ordres de braquages antissymétriques $\delta p \pm \delta q$ à partir des ordres de gauchissement et de tangage.
Les chaînes électriques fournissent les ordres de commande directement aux servovalves électro-hydrauliques contrôlant les distributeurs des servocommandes de puissance (les leviers d'attaque mécanique des servocommandes étant alors débrayés hydrauliquement).
Les actions des dispositifs automatiques s'exercent à divers niveaux. Le trim reçoit, outre les ordres du pilote ceux des centrales aérodynamiques pour assurer la fonction du trim automatique d'incidence et de Mach. Les autostabilisateurs agissent sur les amplificateurs des chaînes électriques.

3.1.2 - Dans le mode de secours de commande manuelle mécanique, les actions du pilote sont transmises par câbles aux servocommandes de puissance par l'intermédiaire de 3 vérins relais (1 par axe), dont l'existence est en fait liée au mode de pilotage automatique. La commande mécanique entre les vérins relais et de puissance est simple du fait de sa nature de secours.
Pour les élevons internes d'une part et les élevons médian et externe d'autre part, un différentiel mécanique réalise les braquages antissymétriques.
Dans ce mode, le seul dispositif automatique disponible, mis à part le dispositif d'effort artificiel, est le trim.

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3.1.3 - Dans le mode de pilotage automatique, les ordres de l'autopilote sont injectés aux servovalves électrohydrauliques contrôlant les distributeurs des servocommandes relais et sont transmis aux chaînes électriques par l'intermédiaire de bielles à ressort à seuil assurant une limitation d'autorité mécanique par réaction sur le dispositif d'effort artificiel.

Dans ce mode, les organes de pilotage et par suite les synchro-transmetteurs des chaînes électriques sont entraînés par les vérins relais par l'intermédiaire de leur levier d'attaque mécanique bloqué par verrouillage hydraulique et désolidarisé hydrauliquement des distributeurs.

3.2 - Schéma de la commande électrique (Fig. 9) -

3.2.1 - Système de commande -

La commande électrique est doublée et comprend des chaînes dites "bleues" et des chaînes "vertes" liées aux corps hydrauliques des circuits bleu et vert des servocommandes de puissance. Elles comprennent deux chaînes (bleue et verte) de synchro-détection par paire d'élevons et deux chaînes pour le gouvernail de direction. L'une des deux chaînes est rendue active, en mode de commande électrique, par mise en pression des servovalves qu'elle contrôle, au moyen d'électrovannes doublées. Sur chaque servocommande de puissance, à double corps hydraulique, la servovalve active contrôle le distributeur d'un corps, et par une liaison mécanique entre distributeurs, le distributeur de l'autre corps.

L'alimentation électrique des chaînes à partir des batteries de bord s'effectue par deux convertisseurs statiques particuliers de haute fiabilité. La fréquence de 1800 hertz a été choisie d'une part pour éviter les interférences par couplage avec la fréquence normale de 400 hertz des autres circuits électriques, et d'autre part pour optimiser la précision. En particulier le rapport de transformation d'une chaîne CX - CDX - CT est ainsi maintenu constant à $\pm$ 1 6 % près pour des températures variant de - 40 à + 120°C.

3.2.1.1 - Une chaîne électrique (bleue ou verte) du gouvernail de direction comprend un transmetteur CX mesurant le déplacement δr_c des pédales, 2 détecteurs CT liés chacun à une servocommande et mesurant le braquage δr des deux tronçons du gouvernail, 2 amplificateurs et 2 servovalves. Transmetteur et détecteurs sont des transformateurs de coordonnées (resolvers), comportant un rotor à un enroulement et un stator à 2 enroulements en croix. Les stators étant reliés électriquement, l'enroulement du rotor du détecteur délivre un signal mesurant l'écart $\delta r_c - \delta r$. L'amplificateur additionne ce signal (dont le signe est déterminé par la phase) à celui de l'autostabilisateur δr_{ST} et délivre à la servovalve l'ordre d'asservissement $\delta r_c - \delta r + \delta r_{ST}$

3.2.1.2 - Une chaîne d'une paire d'élevons est un peu plus complexe à cause des mélanges des ordres de gauchissement et de tangage. Elle comprend un transmetteur CX mesurant le déplacement δq_c du manche, 2 transmetteurs différentiels CDX mesurant avec des signes opposés le déplacement du volant δp_c. 2 détecteurs CT mesurant les braquages antissymétriques des deux élevons δ_D et δ_G. 2 amplificateurs et 2 servovalves contrô-

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lant les deux élevons. Les transmetteurs différentiels, qui sont des transformateurs de coordonnées (resolvers) comportant un rotor et un stator à 2 enroulements en croix, sont intercalés électriquement entre le transmetteur et les détecteurs, de sorte que ceux-ci délivrent des signaux mesurant les écarts $\delta q_c + \delta p_c - \delta_G$ et $\delta q_c - \delta p_c - \delta_D$. Les amplificateurs additionnent ces signaux à ceux des autostabilisateurs $\delta q_{ST} \pm \delta p_{ST}$ et délivrent aux servovalves les ordres d'asservissement

$$\delta q_c + \delta p_c + \delta q_{ST} + \delta p_{ST} - \delta_G$$

et $$\delta q_c - \delta p_c + \delta q_{ST} - \delta p_{ST} - \delta_D$$

3.2.2 - Système de surveillance et de commutations -

La surveillance des chaînes électriques s'effectue par des chaînes de synchrodétection de comparaison, dont les signaux d'erreur sont comparés à des seuils fixés pour déclencher les commutations de mode.

3.2.2.1 - Les 2 gouvernails de direction sont simultanément auto-surveillés par 2 chaînes de surveillance bleue et verte surveillant chacune les 2 chaînes de commande bleu ou verte (Fig. 10). Une chaîne de surveillance comprend un transmetteur CX mesurant l'ordre des pédales δr_c, 2 détecteurs CT mesurant le braquage de chaque gouvernail δr, 2 détecteurs linéaires mesurant le déplacement des distributeurs bleus ou verts des 2 gouvernails, et un comparateur doublé. Celui-ci reçoit les signaux d'écart $\delta r_c - \delta r$ et les corrige par les signaux détectant l'ouverture des distributeurs, de façon qu'un écart dû à la saturation en effort ou au trainage de l'une des servocommandes ne soit pas considéré comme dû à une erreur de commande, dans la limite d'un écart de 2°. Outre cette fonction de correction, les détecteurs linéaires ont pour rôle une détection des embarquements de gouvernails en avance par rapport aux braquages, grâce au fait que les distributeurs commandent en vitesse les vérins. Les signaux résultant sont comparés à un seuil pour commander un relais déclenchant les commutations simultanément sur les 2 gouvernails. La valeur du seuil est fixée à 2°15' par deux considérations. D'abord le taux de commutations intempestives dû aux tolérances des chaînes de commande en fonctionnement dynamique doit être très faible (On estime que l'écart type de 2 chaînes de commande est de l'ordre de 20'). D'autre part l'embarquement dynamique d'une gouverne doit rester compatible avec la résistance structurale, compte tenu du retard des commutations. Par ailleurs des précautions sont prises pour déceler une coupure d'alimentation.

3.2.2.2 - Les élevons internes sont intersurveillés par 2 chaînes de synchrodétection bleue et verte surveillant respectivement les chaînes de commande bleues et vertes. Une chaîne comprend un transmetteur de vitesse 2 mesurant le double de l'ordre du volant $2\,\delta p_c$, un transmetteur différentiel CDX mesurant le braquage d'un élevon $(\delta p + \delta q)_G$, un détecteur mesurant le braquage de l'élevon symétrique $(\delta q - \delta p)_D$, 2 détecteurs linéaires mesurant les déplacements des distributeurs, et un comparateur doublé. Ces

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capteurs sont reliés électriquement de façon à délivrer un signal

$$2\delta p_c + (\delta q - \delta p)_D - (\delta q + \delta p)_G,$$

qui est nul si l'écart des braquages est égal au double de l'ordre de gauchissement, ce qui serait normal en l'absence d'ordres de stabilisation. Ce signal est donc corrigé du double du signal de stabilisation en roulis :

$$2\delta p_c + (\delta q - \delta p)_D - (\delta q + \delta p)_G + 2\,\delta p_{ST}$$

Il est ensuite corrigé par les signaux des détecteurs linéaires de l'erreur de trainage des vérins lorsqu'ils ont des mouvements antisymétriques. Ce signal est finalement comparé au seuil fixé pour provoquer les commutations successives des modes électriques bleu et vert et du mode mécanique.

3.2.2.3 - Les élevons médians et externes d'un même côté de la voilure sont intersurveillés par 2 chaines de synchrodétection affectées aux chaînes de commande bleues et vertes. Mais les commutations de mode successives affectent l'ensemble des élevons, les 2 chaînes symétriques de même couleur étant traitées par le même comparateur.
Une chaîne comprend un transmetteur CX mesurant le braquage d'un élevon, un détecteur CT mesurant le braquage de l'autre élevon du même côté, 2 détecteurs linéaires mesurant les déplacements des 2 distributeurs et un comparateur doublé. Le signal d'écart des braquages corrigé des écarts d'ouverture des distributeurs est comparé à un seuil pour provoquer les commutations successives de modes.

3.3 - Servocommandes de puissance -

La technologie générale des servocommandes de puissance résulte principalement de l'organisation des circuits hydrauliques, de la conception de la commande électrique, et de considérations structurales.

Les circuits hydrauliques comportent deux circuits normaux indépendants dits "bleu et vert". Un circuit de secours dit "jaune" peut alimenter l'un ou l'autre des circuits normaux en cas de défaillance de leur génération. (Cette disposition est considérée comme devant rendre improbable une panne hydraulique totale). D'autre part, la commande électrique des vérins de puissance est doublée par suite des considérations de fiabilité et de performances indiquées précédemment. Dans ces conditions, les vérins de puissance de chaque tronçon de gouverne sont doublés. Ils sont disposés en tandem plutôt que juxtaposés, pour constituer des ensembles intégrés afin de minimiser leur masse et leur encombrement et faciliter leur synchronisation en mode électrique, nécessaire à leurs performances dynamiques.
La sécurité structurale nécessaire pour préserver la sécurité fonctionnelle d'un tel ensemble doublé est assurée par la redondance des attaches sur la structure et des liaisons avec la gouverne : le corps du vérin, comprenant deux corps hydrauliques bleu et vert accolés en opposition à un corps central se déplace le long d'une tige ancrée à ses deux extrémités sur la structure support, et transmet son mouvement à la gouverne par une double liaison. Une glissière assure un guidage approximatif en cas de rupture d'un ancrage de

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la tige. D'autre part l'ensemble est rendu indépendant des déformations élastiques et thermiques de la structure support, par coupure de la tige.

Les connexions hydrauliques s'effectuent par des tubes télescopiques, de façon à éviter les tuyauteries souples, dont l'encombrement serait excessif et la fiabilité douteuse.

Chaque corps hydraulique est constitué d'une enveloppe mince cylindrique en acier chromé coulissant sur un piston porté par un élément de tige. Il porte un bloc de distribution en alliage d'aluminium comportant une servovalve électrohydraulique à retour d'asservissement mécanique, mise en pression par une électrovanne à enroulement doublé, et actionnant un distributeur principal à tiroir sans recouvrement et à gain élevé, un capteur de déplacement linéaire utilisé pour la surveillance électrique et un clapet automatique de by-pass et d'amortissement.

Les deux corps sont synchronisés par la liaison mécanique de leurs distributeurs principaux. Cette liaison est définie de façon à minimiser les effets des dilatations thermiques. En mode mécanique, le levier d'asservissement contrôle directement les distributeurs par cette liaison. En mode électrique ce levier est débrayé hydrauliquement et le contrôle des distributeurs s'effectue par l'une ou l'autre servovalve électrohydraulique. (Le fonctionnement en duplex de la commande électrique entraînant simultanément les servovalves n'a pas été considéré comme praticable à cause de l'influence des tolérances des chaînes électrohydrauliques sur les performances dynamiques de l'ensemble, compte tenu des gains en débits élevés des distributeurs). L'activation de l'une ou l'autre servovalve s'effectue par l'ouverture de l'électrovanne correspondante, commandée par le système de surveillance et de commutation de modes.

L'asservissement en mode électrique est réalisé au moyen de synchrodétecteurs (bleu et vert) montés sur le corps et entraînés en rotation par une biellette redondante (fail safe) dont une extrêmité est articulée sur la structure support. Ces synchrodétecteurs sont enfermés dans un boîtier qui contient aussi les capteurs de surveillance et de répétition de braquage.

Les performances statiques des servocommandes sont remarquables. Leur résolution correspond à 1 minute d'angle de gouverne, leur hystérésis à moins de $\pm$ 1,5 minute.

Les performances dynamiques sont compatibles avec celles des autostabilisateurs. La fréquence de coupure est supérieure à 10 hertz. Aux fréquences des mouvements propres de l'avion et pour de très petits braquages de gouverne, le déphasage n'excède pas 20°.

Ces résultats sont obtenus grâce aux performances élevées des servovalves, à l'absence de fuite statique et à une technologie de distributeurs hydrauliques sans recouvrement.

Les caractéristiques de la rigidité d'ancrage, de l'inertie entraînée et de l'effort nominal sont telles que les servocommandes doivent être stabilisées. Chaque corps possède donc un réseau hydraulique (filtre passe-haut) créant une fuite dynamique stabilisante entre les chambres.

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3.4 - Conceptions et technologie électronique -

Mentionnons seulement quelques aspects typiques communs aux systèmes automatiques de pilotage.

3.4.1 - Duplication et surveillance -

Pour répondre aux objectifs fixés de fiabilité, l'ensemble des systèmes automatiques est redondant. Deux solutions étaient possibles pour réaliser cette redondance avec la surveillance adéquate : solution triplex constituée de 3 chaînes de calcul simples fournissant par comparaison une information réputée sûre, ou solution doublée et autosurveillée avec commutation automatique. Une comparaison des deux solutions serait futile. L'adoption de la dernière solution ne résulte pas d'un choix doctrinal. mais d'une adaptation appropriée à l'organisation doublée des circuits hydrauliques et électriques, entraînant une organisation analogue pour le système de commande de vol normal électrohydraulique.

La redondance exige une ségrégation physique des circuits, qui est réalisée aisément avec des chaînes doublées. Mais l'autosurveillance des éléments exige aussi une ségrégation interne des circuits.

Les techniques d'autosurveillance sont diverses. Les circuits de calcul sont entièrement doublés et leurs résultats sont comparés. Les éléments électromécaniques sont surveillés par comparaison avec un modèle électronique très simple. Une servocommande relais est par exemple représentée par un circuit du premier ordre. Les intégrateurs sont surveillés par comparaison de leur entrée et de leur sortie dérivée. Etc.... Dans tous les cas les comparateurs sont doublés. De plus les surveillances sont fractionnées de façon à ce que les seuils, déterminés par les tolérances de fonctionnement, soient compatibles avec la précision requise, sans que la multiplicité des surveillances partielles puisse nuire à la fiabilité générale.

3.4.2 - Technologie -

L'utilisation des micro-circuits intégrés est généralisée, l'option pour cette technologie ayant été prise dès 1963, pour des raisons de fiabilité et de masse.

Dans les circuits de calcul, les amplificateurs opérationnels du type μA 709 ou dérivés permettent de réaliser de nombreuses fonctions de transfert linéaires et non linéaires par association à des composants isolés. Tous les calculs sont effectués en courant continu. Certains circuits hybrides spécialisés sont aussi utilisés, mais en nombre restreint à cause de leur prix. Des circuits digitaux sont encore utilisés pour réaliser des intégrations à long terme. Les intégrations électro-mécaniques sont proscrites.

Dans les circuits logiques des éléments DTL (Diode Transistor Logic) et même MSI (Medium Scale Integration) sont utilisés. Les relais sont presque éliminés.

CONCLUSION -

Le système de commandes de vol de CONCORDE est conçu pour répondre à des objectifs de performances propres à sa mission supersonique et des objectifs de fiabilité propres à son utilisation opérationnelle civile. Une commande électrique, inspirée de réalisations militaires avancées apporte la précision requise et facilite l'intégration des aides automatiques de pilotage. Ce système reste cependant conventionnel par certains aspects (organes de pilotage, liaison pilote - gouvernes univoque, limitations d'autorité mécaniques, etc...) du fait des contraintes imposées par les habitudes de pilotage et les garanties de sécurité. CONCORDE, premier avion de transport civil à utiliser une transmission électrique des ordres de pilotage intégrant un ensemble d'aides automatiques, ouvre la voie aux conceptions de pilotage nouvelles appropriées au transport aéronautique de l'avenir.

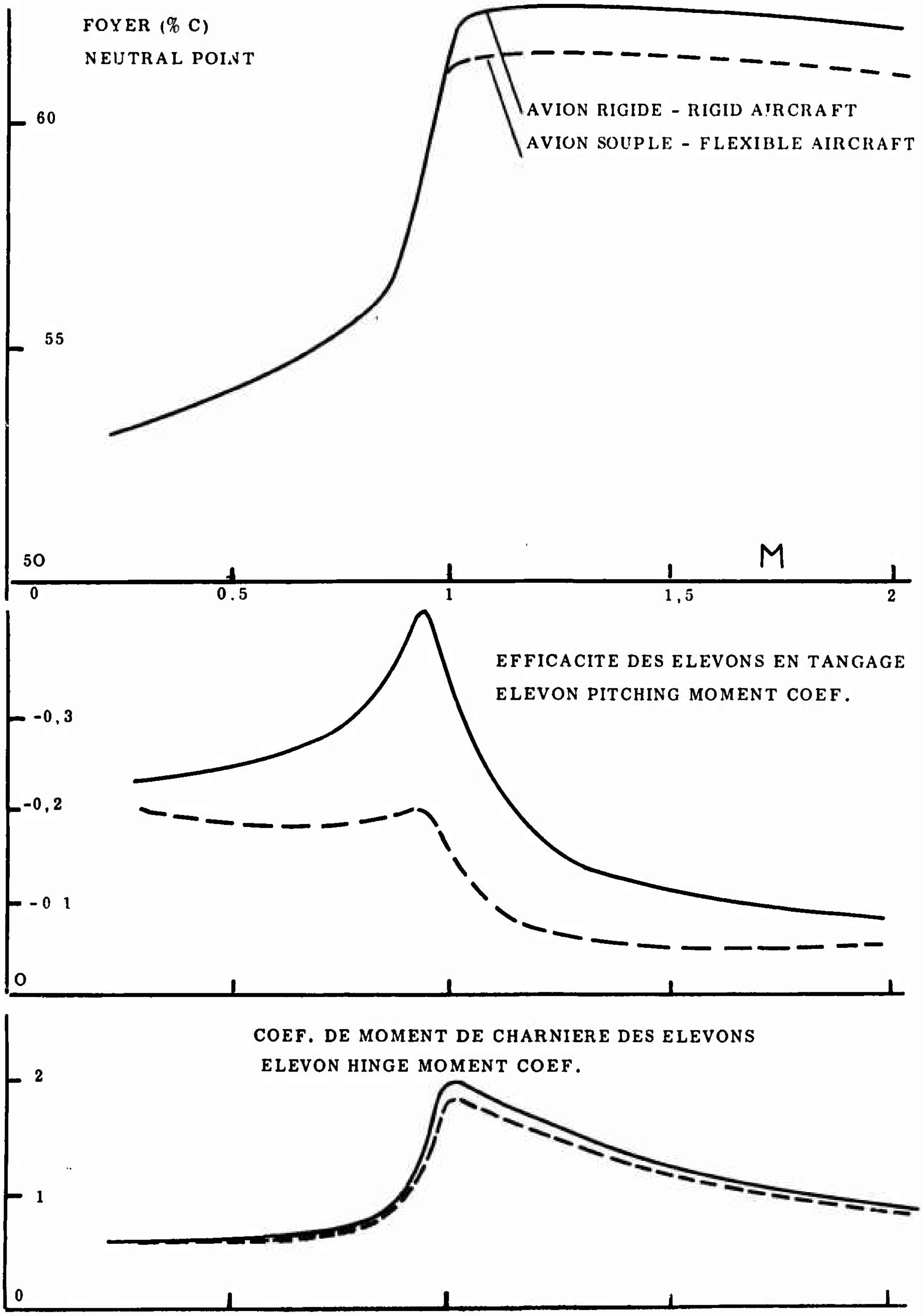


FIG. 1

TSS - PARAMETRES DE LA MANOEUVRABILITE LONGITUDINALE
LONGITUDINAL MANOEUVRABILITY PARAMETERS

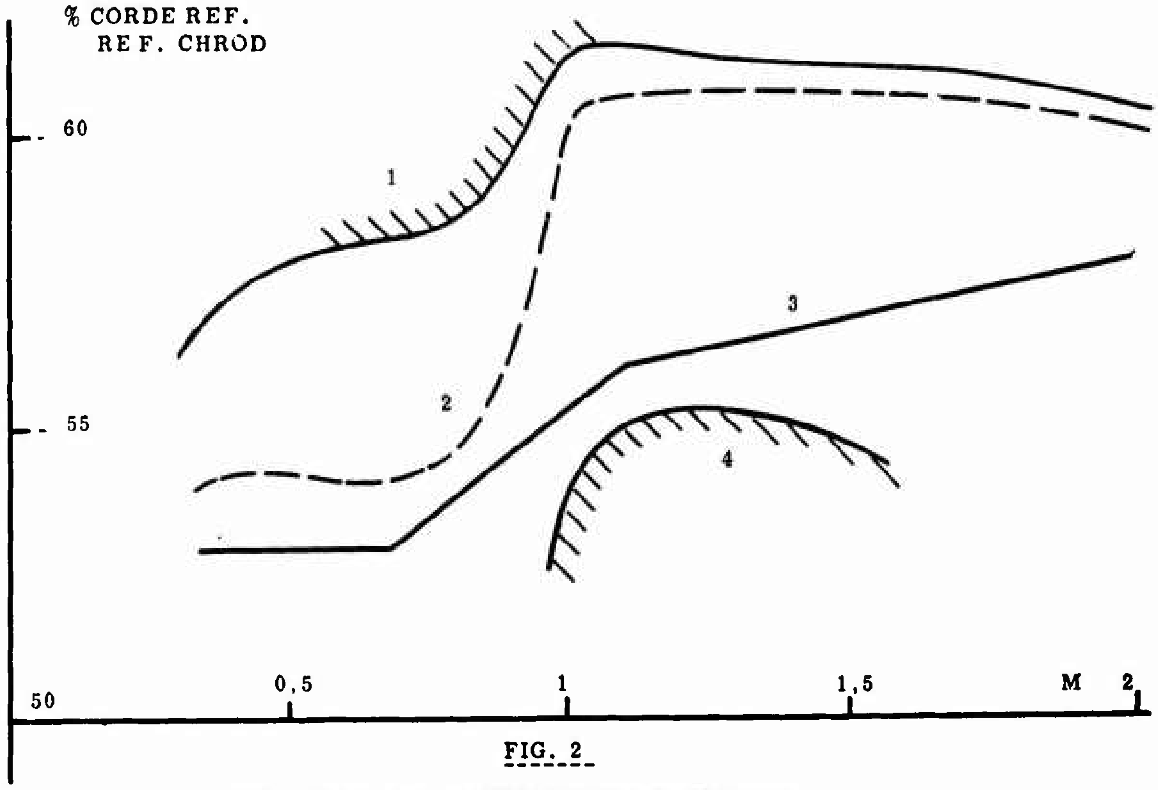


FIG. 2

TSS - COULOIR TRANSSONIQUE EN MONTEE
TRANSONIC CORRIDOR IN CLIMB

1 - LIMITE DE MANIABILITE - MANIABILITY LIMIT
2 - ID. SANS AUTOSTAB. - ID. WITHOUT AUTOSTAB.
3 - CENTRAGE MOYEN - AVERAGE C.G. POSITION
4 - LIMITE DE MANOEUVRABILITE - MANOEUVRABILITY LIMIT
(N = 1,3 - PANNE HYD.) - (N = 1,3 - HYDRAULIC FAILURE)

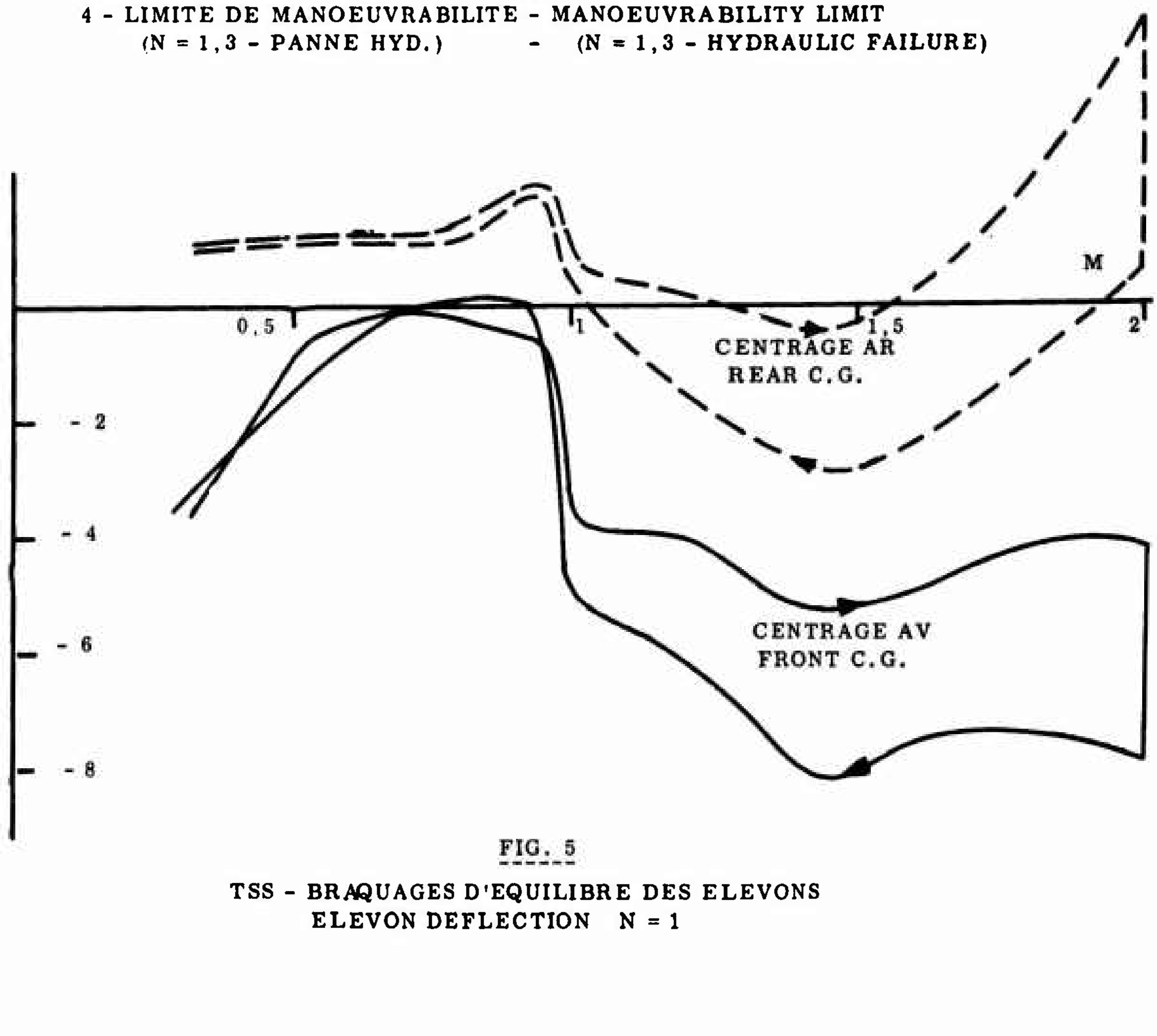


FIG. 5

TSS - BRAQUAGES D'EQUILIBRE DES ELEVONS
ELEVON DEFLECTION N = 1

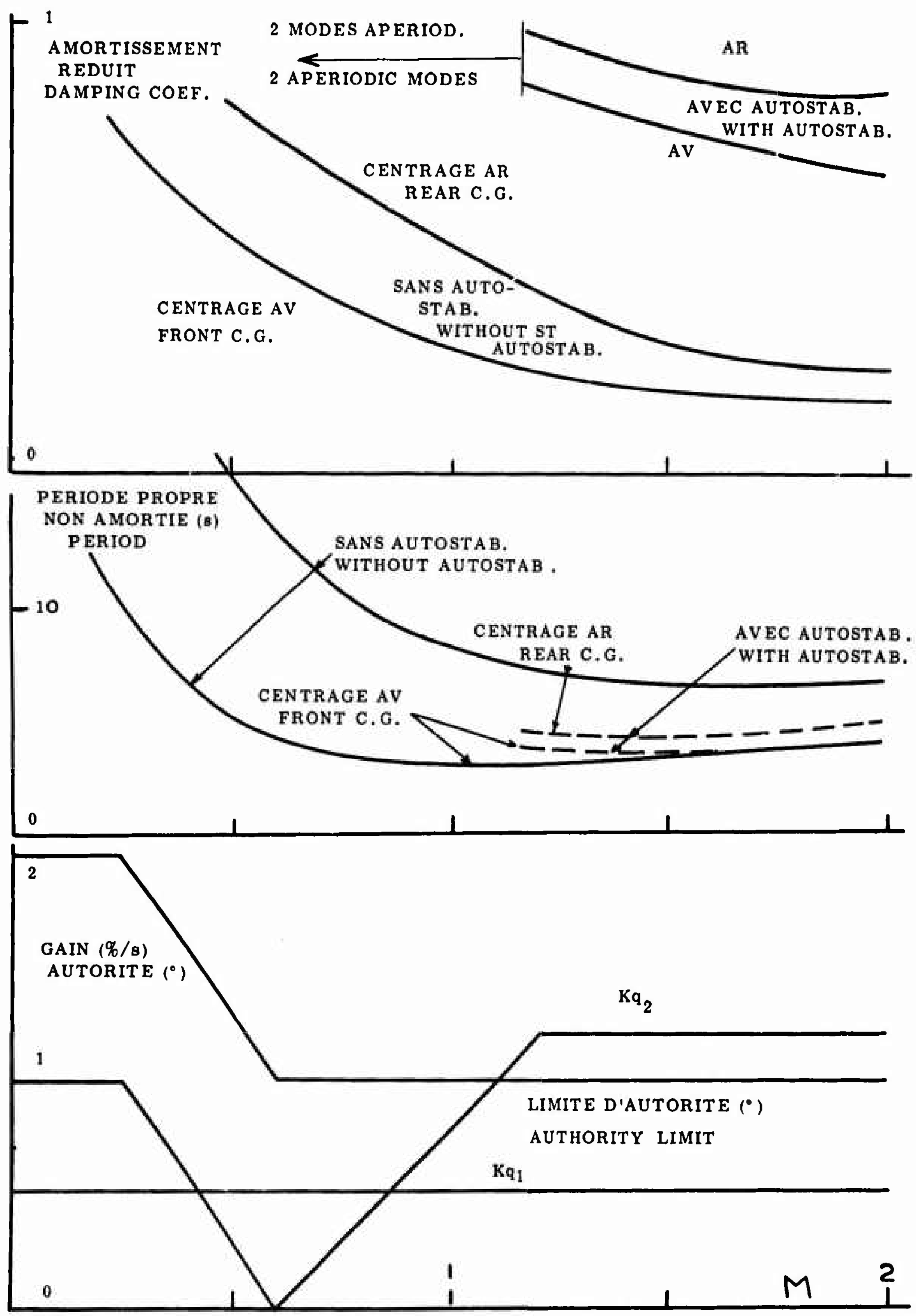
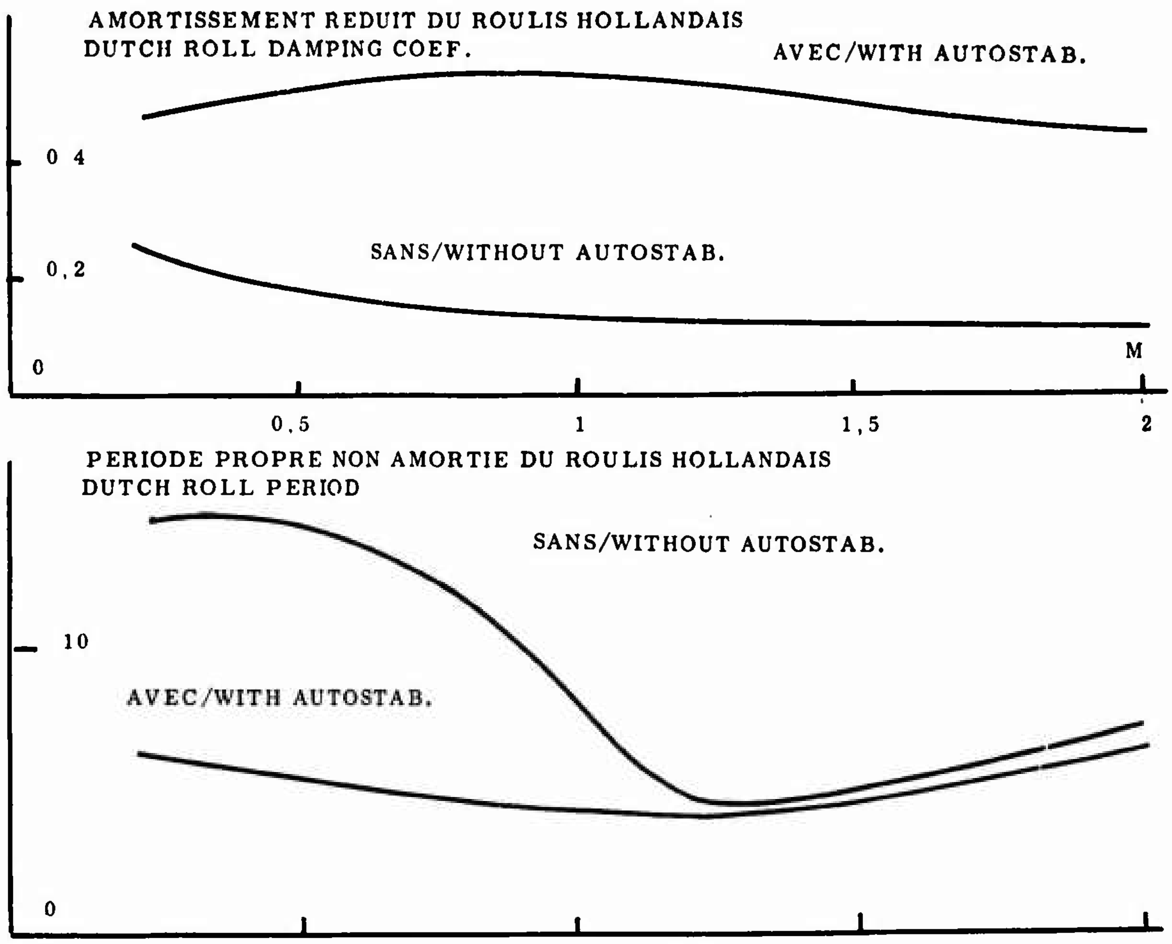
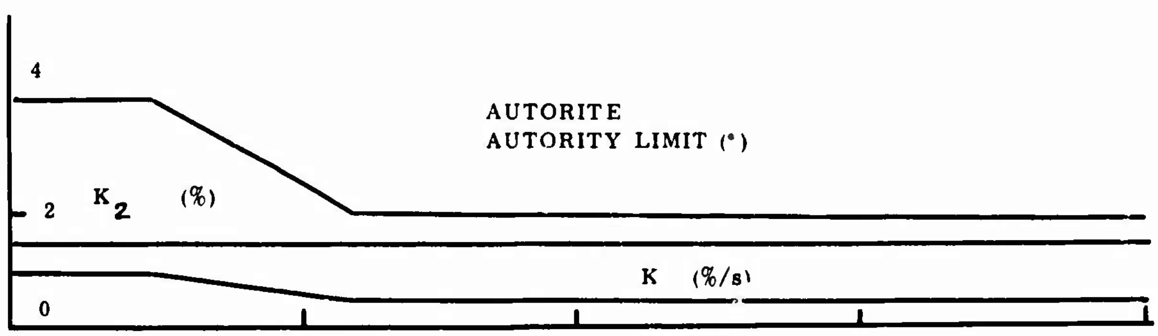


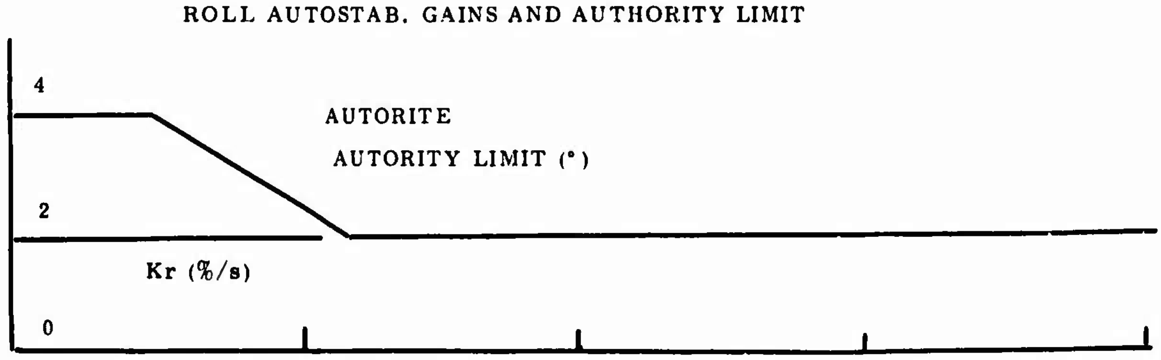
FIG. 3

TSS - AUTOSTABILISATEURS DE TANGAGE





GAINS ET AUTORITE DU STABILISATEUR DE ROULIS
ROLL AUTOSTAB. GAINS AND AUTHORITY LIMIT



GAIN ET AUTORITE DU STABILISATEUR DE LACET
YAW AUTOSTAB. GAIN AND AUTHORITY LIMIT

FIG. 4

TSS - AUTOSTABILISATEURS TRANSVERSAUX
LATERAL AUTOSTAB . SYSTEM

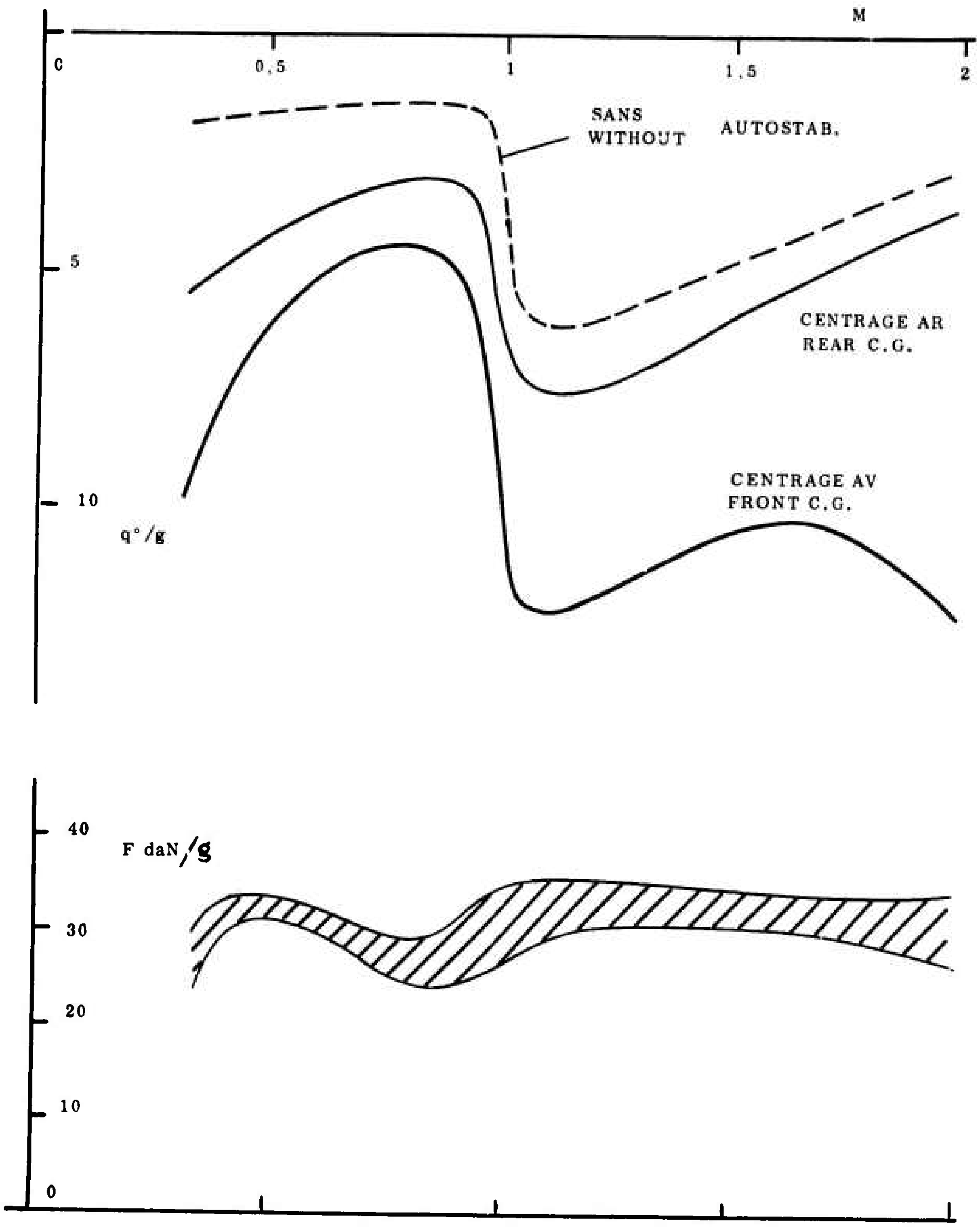


FIG. 6

TSS - BRAQUAGES ET EFFORTS/G EN VIRAGE N = 1,5

ELEVON DEFLECTIONS AND FORCE/G IN TURN N = 1,5

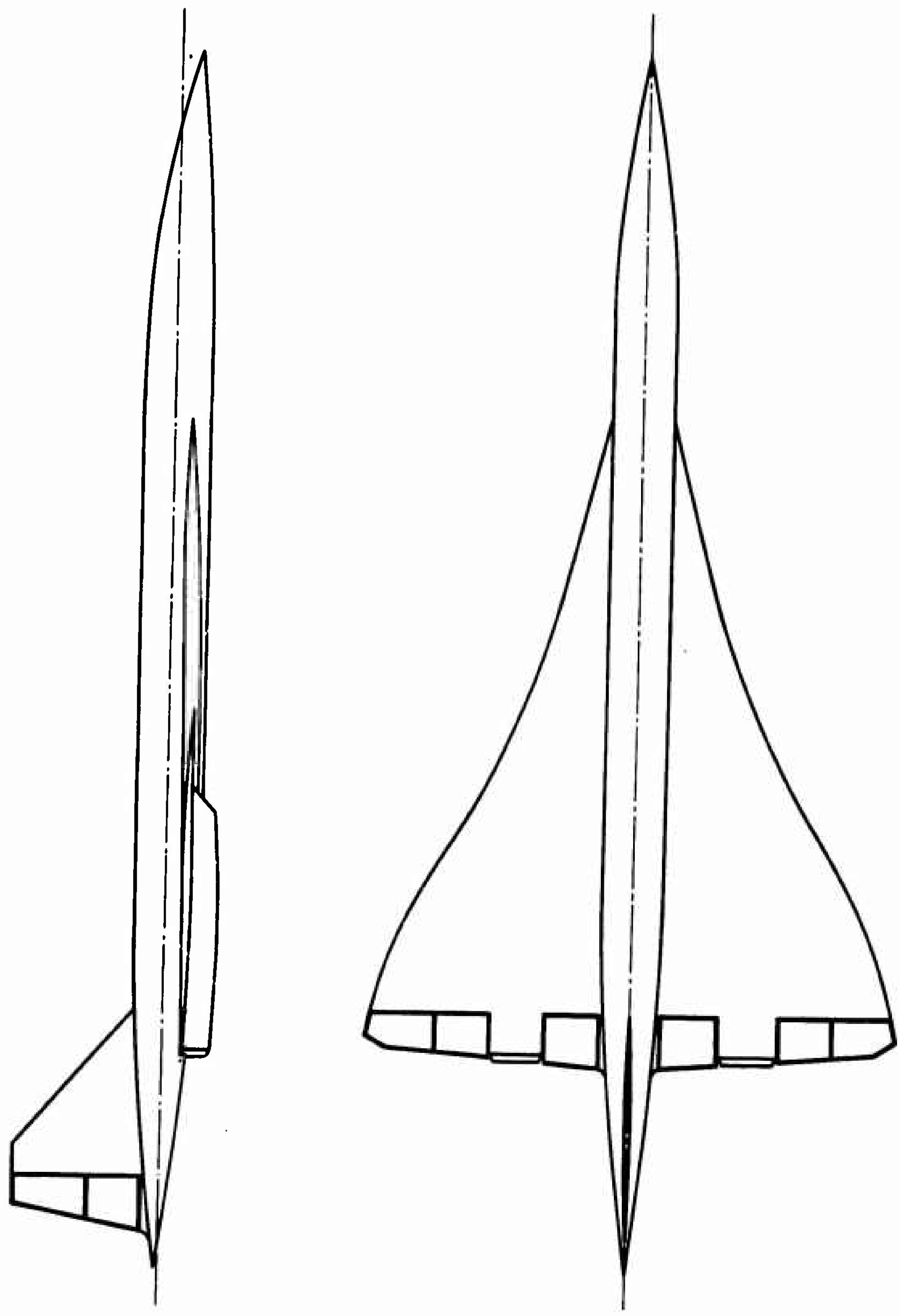


FIG. 7

TSS - GOUVERNES
CONTROL SURFACES

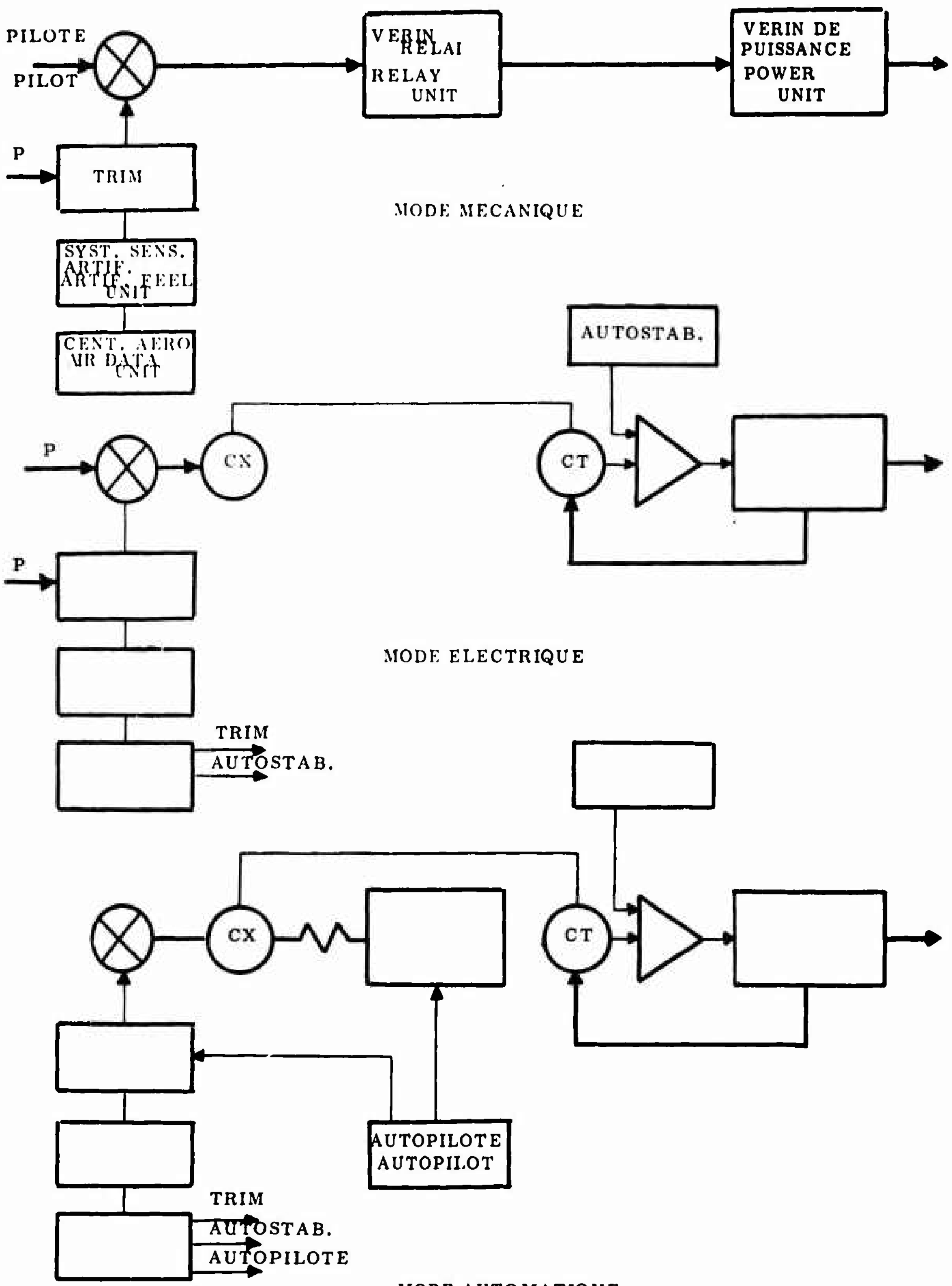


FIG. 8

TSS - MODES DE PILOTAGE

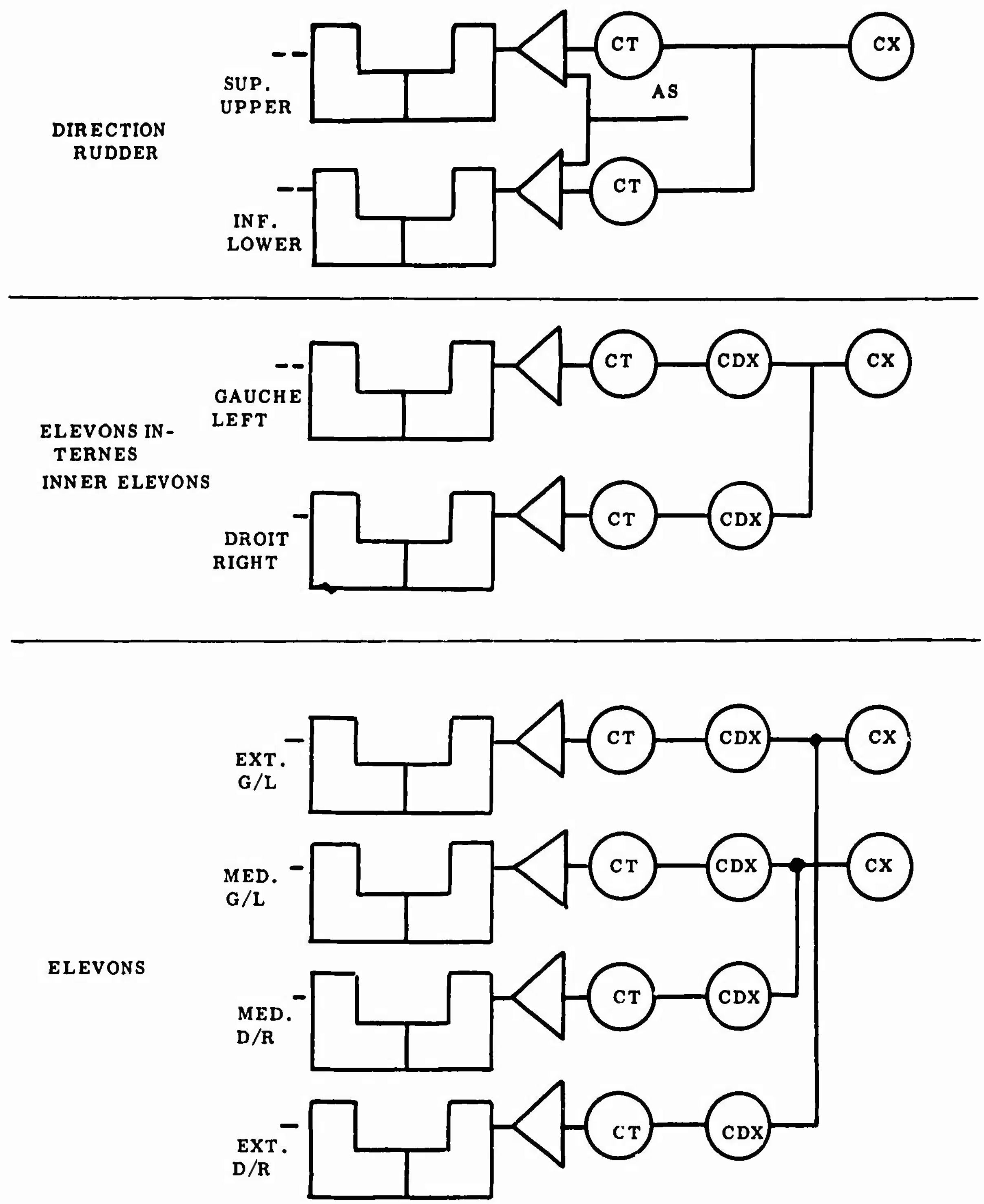


FIG. 9

TSS - COMMANDE ELECTRIQUE (CHAINE BLEUE)
ELECTRICAL SIGNALING SYSTEM (BLUE CHANNEL)

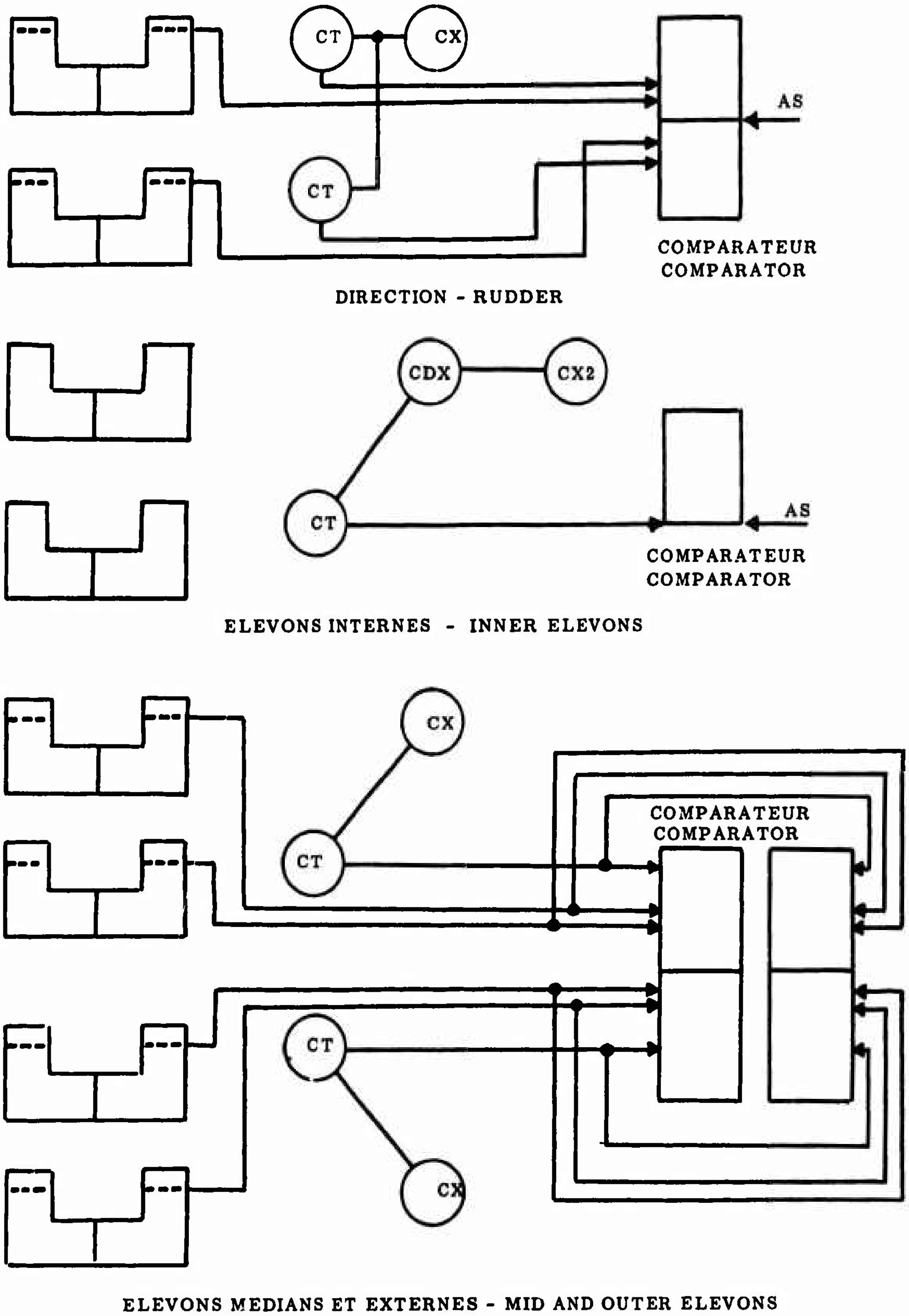


FIG 10

TSS - SURVEILLANCE DE LA COMMANDE ELECTRIQUE BLEUE
MONITORING SYSTEM OF THE BLUE CHANNELS

ADVANCES IN AIRCRAFT CONTROL SYSTEMS WITH PARTICULAR REFERENCE TO COMBAT AIRCRAFT

by

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Abstract

A review of current flight control system design for Combat Aircraft is given highlighting the reliance already placed on forms of electrical signalling of the flying control surfaces and the increasing use of feedback control techniques to achieve satisfactory handling qualities. In all current systems, however, a mechanical backup system is retained.

A description of a possible electrical signalling system design is given, including manoeuvre demand control characteristics and some of its advantages are discussed.

The conclusion is reached that, although some experience is being gained in service of forms of electrical signalling, aircraft designers have not yet the confidence to eliminate mechanical reversion systems and these, unfortunately, often compromise the primary electrical signalling system performance. Recent system developments should lead to the abandoning of these mechanical reversion systems in future project designs and the full benefits of feedback control can then be obtained. These include the optimisation of the overall airframe taking advantage of feedback control and new cockpit layouts taking advantage of the use of small side controllers.

1 INTRODUCTION

In the past, automatic control has been considered in aircraft designs as a means of supplementing the basic aircraft stability and handling qualities. A great deal of effort has been put into the aerodynamic design to achieve good performance from the aircraft with a minimum loss of stability and control characteristics. However, it is quite clear that each successive generation of aircraft has relied more on automatic control than the previous generation (see Figure 1) and yet the automatic control system is often looked upon by many aircraft designers as an unavoidable nuisance which makes the aircraft more complex and need more maintenance. Because of this basic distrust in 'black boxes', the control system has had many constraints placed upon it, such as authority limitation, which has in turn wasted some of the development potential since too great a change in dynamic characteristics would not be permissible when the system saturated or when the system failed. The recent developments using 'failure survival' principles have gone some way to increasing the effectiveness of automatic control systems but reversion or 'get you home' systems are retained for safety. These latter systems are now presenting severe design difficulties since, if the aircraft was acceptable with such a simpler reversion system there would have been no need for a more sophisticated primary system. In practice, the reversion system may be acceptable over a limited flight envelope which would enable the aircraft to be returned to some diversionary airfield.

In current project designs for Advanced Combat Aircraft, there is a great deal of ingenuity being employed to produce schemes for a high performance primary control system using electrical signalling and feedback control but with a mechanical reversion system to cater for certain double failure cases. However, in a great deal of cases, the very presence of the mechanical reversion system degrades the performance of the primary system and can add to the complexity because of the need to synchronise the mechanical reversion. This paper discusses some of the problems that can be expected in some of the proposed schemes and compares these with a full "fly-by-wire" system with no compromise of mechanical reversion. The development time-scale is also included as a major constraint in the ultimate choice of system design. Some possibilities of future developments are also given taking advantage of electrical signalling with no mechanical reversion.

2 TYPICAL CURRENT SYSTEM DESIGNS

Each aircraft designer has his own individual ideas of flight control system design and consequently there exists today a whole range of system designs but, in order to achieve high performance from the autopilot and to achieve good autostabilisation in regions in the flight envelope of high control sensitivity, some form of 'electric signalling' has been introduced as the primary mode. There follows a brief description of six main types of flight control system currently in use or being considered in project designs with an indication of the problems in integrating the subsystems and providing mechanical reversion. As an example, the pitch control system is considered, but similar problems exist in the lateral and directional controls.

2.1 System with limited authority series actuators for autostabilisation and full authority electric operation of main power controls for autopilot modes

Fig. 2 shows a simplified block diagram of this system. The main subsystems that are of real importance are:-

(a) Pilot's mechanical controls
(b) Artificial feel system
(c) Autostabilisation system
(d) Autopilot system
(e) Air Data Computer system
(f) Trim system

For convenience, the discussion of the system operation is divided into two parts relating to manual control and fully automatic control.

2.1.1 Manual control

The artificial feel system is programmed by signals from the Air Data computer in order to achieve a satisfactory relationship between stick force and aircraft response (for instance stick force/g within a certain range). The control stick movements per unit response will of course be variable and in order to maintain good resolution with high control sensitivity, the total stick travel needs to be of the order of 12 inches - this being determined by the practical resolution of mechanical control runs. Because the stability of the aircraft needs augmenting in some parts of the flight envelope, an autostabiliser system is incorporated using series actuators in the control runs. The two main problems that arise are firstly the mechanical engineering aspect of ensuring that the impedance of the mechanical runs ahead of the actuators is very large compared with that part downstream including the valve forces of the main power controls. In order to overcome this difficulty, various means have been used such as extra servos in the linkage to form 'irreversible' points or two-stage valves to reduce friction and Bernouilli force effects. The second problem is one of deciding on the actual value of the authority limits. If the transient and steady state responses are changed by a large amount by the introduction of the autostability terms, then not only will the response be non-linear with amplitude, due to system saturation, but also the artificial feel system parameters will have to be set to some compromise value between the two cases of autostabiliser on and off. Quite often, the limits have had to be varied with flight conditions (from Air Data Computer) in order to have sufficient authority at low speed while maintaining safety at high speed. Recently, multiplex or duplicate monitored system have been used in which the actuators have larger authorities, thus reducing these integration problems.

However the mechanical system still limits the overall performance and to a certain extent the whole suffers from the worst features of both mechanical and complex redundant electronic systems. It has been suggested in one recent application that a self-adaptive multiplex autostabiliser system tends to adapt against the effects of the feed forward control term given by the mechanical control run rather than to the changing dynamic characteristics of the aircraft.

2.1.2 Autopilot control

In this design, the autopilot is integrated with the basic flying control system through the main power controls. In the autopilot mode, the mechanical feedback is replaced by an electrical feedback and the small autostabiliser actuator controls the main valve. In principle, full authority is available but in practice the mechanical pilot's linkage is fixed to the output of the power control and drives the pilot's controls and artificial feel system. This latter enables an authority limit to be placed on the autopilot corresponding to a given level of artificial feel force. Thus, if a constant stick force/g was designed into the feel system, then the autopilot limitation would be satisfactory. In fact this facility is one of the main design constraints on the feel system scheduling and it has not always been possible to achieve satisfactory autopilot limitation by this means and some extra response monitoring (using normal acceleration, pitch rate and incidence) is then employed.

The performance of the autopilot is capable of being set to a high value since the servo arrangement has high resolution and good frequency response. In fact in one project design, a high gain pitch rate demand 'inner loop' control system was used to obtain very high performance terrain following. A further benefit of the manoeuvre demand system is a simpler manoeuvre limitation arrangement.

2.2 System using electric signalling of powered controls with mechanical standby

2.2.1 Manual control

Fig. 3 shows a simplified diagram of the system. The primary electric system allows the integration of the autostabilisation without the mechanical engineering problems of the system described above in section 2.1. However, the authority of the autostabiliser is limited in this design by the amount of allowable 'lost motion' in the mechanical circuit. The artificial feel system still has to provide the main contribution to meet the requirement of stick force/g and so the problems of non-linear response due to automatic control saturation are again present and will limit the allowable amount of feedback control. Also the artificial feel system is still complex and a good mechanical control run is still required as a reversionary mode.

2.2.2 Autopilot control

In this system, the autopilot is not integrated into the primary flying controls and in fact requires a separate electro-hydraulic actuator close to the feel system to drive the stick and hence (through the electrical signalling system) the control surfaces. If the integrity required of the autopilot is an order or more below that of the primary flying controls, then it is logical to retain this complete separation, but the penalty is paid in performance and in the added complexity of having the extra servos. Furthermore if automatic landing or terrain following, etc., of high integrity and reliability is required, then these extra actuators need to be of a redundant form and in practice involves an inordinate increase in components in the overall system. The operational advantages claimed are twofold: the feel system allows torque limiting to be used thus limiting the manoeuvring authority of the autopilot and the stick follows autopilot demands (which is a current requirement in some aircraft types). However, the former point is debatable since there remains the difficulty of providing a feel system giving constant stick force/g and the latter requirement may well be relaxed in the future since it is not necessarily the only way to achieve the real aim of smooth and safe transition from autopilot to manual control. The automatic trim system is in operation during autopilot operation in order that the manual reversion is made in the safest way but involves the addition of another subsystems which, in the case of a 'failure survival' autopilot also has to have redundancy in order not to reduce the system integrity if manual reversion is to be considered in a critical flight condition.

2.3 System with larger authority multiple series actuators (Fig. 4 refers)

2.3.1 Manual control

The primary control depends upon a multiplex manoeuvre demand or "command augmentation" system operating through a triplex actuator system giving a single failure survival capability. The control law for a high performance aircraft is based on pitch rate and normal acceleration feedback and a simple feel system only is available and so devices such as bob weights are required to give some partial mechanical feedback of normal acceleration. A series trim is required to synchronise the mechanical reversion. Even allowing for these mechanical devices, the flight envelope is still usually limited for which acceptable handling qualities are obtained using the reversion system. Another variation of this system is to employ a conventional programmed artificial feel system and stick force sensors to provide the electrical command signals. Such a system has better reversionary characteristics and will be discussed in para 2.4 below.

2.3.2 Autopilot control

With this type of primary control system it is possible to have two autopilot configurations. One type would use parallel actuators driving the pilot's stick and hence operate through the manoeuvre

demand system. (This system is shown in Fig. 3). Another method, which at first sight appears to be a more integrated solution, is to feed the autopilot demand directly into the multiplex manoeuvre demand system. However, due to the large authority required if the autopilot is required to be in operation during transition from high level supersonic speeds to low level subsonic speeds, a slow acting series trim actuator is necessary thus allowing the multiplex actuators to operate within their limited authority.

In both arrangements, the autopilot design is simpler than in the former arrangements since the manoeuvre demand system provides a high degree of autostabilisation against turbulence and the flight path control terms provided by the guidance system are more straightforward. Also the safety of the structure can be assured by manoeuvre demand limitation thus overcoming the problems in the previous systems or providing accurate feel system open-loop programming.

2.4 Multiplex Manoeuvre Demand System (Series Actuators) with Mechanical Reversion

Figure 5(a) shows the main features of the system. Because the electric signal from the stick force sensor tends to zero in the stick trimmed condition, the surface position to trim has to be provided either by the mechanical input or an integrator in the controller C(s). The latter scheme, although giving optimum performance, has two disadvantages in practice. Firstly the triplex actuator authority would need to be large or an autotrim system will need to be employed using a logic system such that the trimming only occurs when the stick force is zero thus preventing the trim system operating during manoeuvres. Secondly the reversion to mechanical would cause a large transient due to the lack of synchronisation of trim and again the autotrim would be necessary. If the feel system has suitable programming from the Air Data Computer (A.D.C.) and wing sweep (if appropriate) then the steady manoeuvre characteristics of the mechanical reversion would probably be acceptable over a large part of the flight envelope although the transient responses to the steady state may not be good without the command augmentation system. Thus the compatibility of the primary C.A.S. and mechanical reversion can be achieved by the use of series actuators, programmed feel and autotrim. However, the series actuator and hence primary electric control system relies on the mechanical control system having ideal characteristics such as no backlash, low valve forces, etc.

2.5 Multiplex Manoeuvre Demand System (Series Actuators with electrical feedback from main P.F.C.'s) with Mechanical Reversion

Fig. 5(b) gives the essentials of the scheme which only differs from that in 2.4 above by including an electrical feedback from the main jack and including the movements of the mechanical command and mechanical P.F.C. feedback in the first stage actuator feedback measurement. Another important feature is the stick position pickoff which includes implicitly a stick trim position signal. The principle of this scheme (proposed by B.A.C.) is to synchronise the Manoeuvre Demand System with the Mechanical Reversion and so the multiplex first stage actuator should tend to recentre in the steady 1g flight case. One important advantage of this form of synchronisation is that no clutch is required to re-engage the mechanical reversion which is at all times connected thus increasing the integrity of the reversion. The first stage actuators, however, do need a self centring device when not energised. Also, as in 2.4 above, the primary system performance is conditional upon good characteristics of the mechanical system otherwise feedback of forces to the pilot's control column and stick sensors will occur.

2.6 Multiplex Manoeuvre Demand System with Clutched Mechanical Reversion

Fig. 5(c) shows the main features of this scheme. In order to eliminate the need for a sophisticated form of clutch and disengage mechanism of the mechanical controls on the P.F.C., in order to accommodate a large trim disparity between the electrical signalling and mechanical systems, two electrical inputs are employed. The first is the stick position relative to trim and the second is a trim position signal. This scheme still requires the disconnect mechanism to have a fairly large authority to accommodate the Manoeuvre Demand Control demands relative to the mechanical system demands, but, in 1g flight, the error should tend to zero. It would be possible to add an autorim feature if it was considered desirable as in 2.4 above. The main advantage of the clutched scheme is that in the primary mode, the mechanical control system characteristics do not degrade performance, but of course the clutch mechanism must be designed to have high integrity.

3. AN INTEGRATED MANOEUVRE DEMAND SYSTEM DESIGN

This system is one in which full reliance is placed upon the primary electronic controls ('fly-by-wire') and the system employs fourfold redundancy to achieve the necessary reliability for the complete flight time (double failure survival capability).

3.1 Manual control

The pilot's controller can be either a conventional stick or a small side controller since no mechanical reversion considerations are required. Quadruplex position transducers feed signals to four 'lane packs' of electronics. These are separated by the maximum distance possible, commensurate with installation difficulties, in order to maintain the highest integrity to common failure hazards. The final outputs from the electronic units drive the quadruplex electro-hydraulic actuator units. The output of these are in the form of a single fail-safe consolidated mechanical displacements which in turn controls the position of tandem main power controls with local mechanical feedback. The rate gyro and accelerometer terms are obtained from four of each unit feeding into the electronic units providing some particular manoeuvre demand control law. The feel system consists of a fixed spring system and, depending on the control law, a trim input may be necessary in the form of a datum shift of the input

pick-offs. Gain scheduling may well be required to achieve stability and response requirements over the flight envelope and in Fig. 6 these are shown as being derived from Air Data sources. There is a problem of providing the gain scheduling to a standard of integrity as high as the basic signal chains. As drawn, a simple logic system takes the lower of the two from two self monitored Air Data Computers thus providing a single failure survival capability with a standby fixed gain for the second A.D.C. failure case. The performance with a standby fixed gain would not be optimum, but due to the basic use of feedback, the performance would be acceptable. With this failure condition, a good deal of the flight instrument and navigation systems would be also in reversionary modes and the aircraft mission may need to be curtailed for other reasons. A simple form of self-adaptive gain changing system may be preferable from the performance point of view and would allow the system to operate without interface with any external system such as an Air Data computer. This argument applies for simple self-adaptive designs in which redundancy can be applied easily.

3.2 Autopilot control

The level of integrity of the autopilot can be chosen freely with this basic system. A full redundant autoland system for scheduled operation would be the ultimate goal and this is compatible with this system. Fig. 4 shows a duplicate monitored autopilot arrangement (single failure survival), but this could be extended to the same fourfold system as the primary manual system. All the points of criticism in the previous system designs are met since full authority is available and manoeuvre demand limitation is provided per se.

4. PERFORMANCE CONSIDERATIONS OF AUTOMATIC FLIGHT CONTROL SYSTEMS

Experience with manoeuvre demand control has been limited to experimental programmes such as the X 15 in U.S. and the Avro 707c in U.K. The flight proving of the F 111 is also relevant since a fairly large authority 'Command Augmentation System' is fitted which gives, within limits, a constant response throughout the flight envelope to pilot's commands. The actual response parameter that is used is one that seems very popular in the U.S. at the moment and is based on some work by Boeings[1]. This parameter, labelled C*, is a blend of normal acceleration at the pilot's station and pitch rate:-

$$C^* = \frac{V_c}{g}\dot{\theta} + [n - 1]_{cg} + \frac{\ell}{g}\ddot{\theta}$$

where n = normal acceleration of cg measures in units of g,

V_c = 'Cross-over velocity' where the $\dot{\theta}$ and (n-1) contributions to C* are equal and

ℓ = distance of pilot ahead of cg

Refs. 1 and 2 discuss various aspects of this criterion and Ref. 3 shows another method of achieving the 'steady state' requirement as a compromise between the high speed manoeuvring requirement of constant stick force per g and a suitable low speed characteristic. Another area that has received recent attention is the use of a prefilter between the pilot's commands and the basic feedback loop. A phase lead filter can be used to partially cancel the effect of a low value of lift slope and high wing loading and so improve the response in normal acceleration to pilot's commands without affecting the good ride qualities of such an aerodynamic configuration in vertical turbulence. Figs. 7a, b, c show at three extreme flight conditions, an aircraft longitudinal response to a step stick command with the system of Fig. 5(a). The parameter plotted is C^*_N, where the 'cross-over velocity' V_c equals 400 ft/sec. It is clear from these figures that the use of a prefilter is a powerful method of improving the response and the very difficult case in Fig. 6c (where the dimensional value of L_α is as low as 0.16 sec^{-1}) could be met if necessary by the rather extreme filter (1 + 6s)/(1 + s).

Two main criticisms have been made of the general handling qualities of pitch rate/normal acceleration longitudinal manoeuvre demand systems. The first is the new piloting technique required because of the neutral 'speed stability'. That is for any change of speed from a trimmed condition, the attitude is held constant with the stick force zero and so the aircraft would not return to the original trimmed speed. It is interesting to note that in Ref. 2 a conclusion was reached that 'near-zero stick force per V was found acceptable by pilots' whereas in Ref. 4, the X 15 team of pilots did not like this characteristic and in fact, for landing, an artificial nose down out of trim was introduced so that the pilot was required to provide a steadily increasing backward stick movement during the landing flare. In the Avro 707c, there was a spread of pilot opinion on this characteristic and most pilots felt that the advantages of the trimming effect of the system outweighed the neutral stick stability. The second criticism of a manoeuvre demand control system is that the position of the aerodynamic controls is not directly related to the position of the cockpit controls[6]. For instance, in a cross wind landing, the pilot's lateral control would be nearly central and yet 90% or more aileron deflection approach could be applied to counteract a steady sideslip. Another example would be the approach to a stall where the system would counteract any pitching moments until the control limits were reached, thus causing a precipitous stall characteristic. However, it is quite simple to provide a visible and/or audible warning when a certain proportion of control authority is applied. Current designs of stall warning and 'stick pusher' systems are based on similar principles.

In order to satisfy more than one performance criteria, more than one control loop is required. For instance, by feeding back some functions of roll rate to the ailerons and a function of yaw rate to the rudder, the two criteria of rolling response and dutch roll damping can be met. This particular problem of optimising the lateral control system shows some typical properties of multiloop systems. In particular, the numerator or 'zeros' of the basic aircraft transfer functions (such as v/ξ) can be modified thus controlling the transient response characteristics as well as the stability. Another well known multiloop system is the combination of elevator and throttle control to produce a

satisfactory height hold autopilot below minimum drag speeds. Recent work has shown that, by suitable design of a multiloop system, the addition of a direct lift control to elevator control can produce more nearly ideal flight path response.

The blending of the various feedback loops is another advantage given by a full electrical signalling system.

5. COMPARISON AND DISCUSSION OF THE VARIOUS SYSTEMS

The main difference between current systems and an integrated manoeuvre demand system is that the former consists of a set of differing subsystems with many interface areas, whereas the latter consists of a simpler set of identical units operating in a parallel redundant signal chain. The comparison can be made under the four headings of performance, safety, reliability and integrity, and serviceability.

5.1 Performance

In manual flying, a manoeuvre demand system has many advantages over a conventional one and these all lead to a reduced pilot workload. Some of the operational aspects are discussed in section 4. For instance, instrument flying should be made easier since a highly stable aircraft is obtained throughout the flight envelope at the same time as providing the optimum response characteristics. The higher frequency 'anticipatory' terms could be eliminated from the director displays thus simplifying the computing and reducing noise on the display. A particular example of a director mode is a take-off law. With present control systems the rotation of a delta aircraft with trailing edge flap controls requires a complex pilot control action to lift the nose wheel, rotate and flare into the climb-out flight path and a pitch rate to speed and acceleration control law is being used. If a pitch rate demand system is employed as the basic manual control, then the required control action will be simply related to the desired pitch rate demanded on the director display - thus reducing considerably pilot workload.

It has been suggested that manoeuvre demand systems are not wanted because a constant response is not required over the flight envelope but that different responses are required. However, by use of built-in highly reliable feedback control, the desired model response at each flight condition is more likely to be achieved. Current hybrid systems employing limited authority electrical systems have reduced performance with subsystem failure. This reduction in performance at best involves a minimum exploitation of feedback control techniques and at worst can cause serious flight limitations after certain single failures and certainly after double failures. Recently, schemes providing single failure survival manoeuvre demand with mechanical reversion (such as in paras. 2.5 and 2.6 above) have been evolved which give a useful reversionary mode by employing conventional programmed feel. These schemes are attractive if the feedback terms can be lost without producing unacceptable handling qualities and the timescale of the programme of development means the use of current "on-the-shelf" hardware. However a quadruplex system is designed to attain the desired performance after two failures and so is superior although the development of such a system is not yet complete. Programmes in the U.K. and in the U.S. are progressing along similar lines to develop quadruplex double failure survival systems.

5.2 Safety

The safety requirements of a flight control system are designed to prevent overstressing of the whole aircraft or any part of the airframe as the result of inadvertent pilot action or as a result of failures in the control system itself. In most current designs, the artificial feel system provides this basic structural safety both in manual and autopilot modes of control. However, this is only possible if the open loop scheduling of the feel force gradient exactly compensates for the aerodynamic and inertial changes of the aircraft throughout the flight to produce a constant longitudinal stick force/g and constant lateral stick force/roll rate. A manoeuvre demand system provides this constant response to stick force and stick movement and so ensures the basic structural safety in the most satisfactory way.

The protection against failures in a system using similar redundancy is obtained by some majority voting technique and the flight path disturbance is minimised. With a response monitor, such as the use of the artificial feel system to limit excess normal acceleration, relatively large flight path deviations are possible after system failure and, although structural safety is assured, these deviations are not acceptable at low altitude. In this latter case a 'fail-soft' system can only be obtained by the use of fairly close tolerance monitoring afforded by multiplex or multiplicated systems using similar redundancy.

5.3 Reliability and integrity

One of the great problems in current systems is to establish the basic integrity since the analysis of failure modes and their consequences involves a study of many different subsystems and the interface areas. In practice the interface between systems is very complex: for instance the Air Data Computer supplies more than one signal to the feel system, autostabiliser, autopilot, autothrottle and autotrim. Also the performance of the overall system deteriorates to a lower level after subsystem failures and it is sometimes very difficult to establish whether the aircraft system is then 'unsatisfactory' or 'unacceptable' and the probabilities of these failures have to be related to the risks produced. This process has reached a very difficult point in some current designs since it is sometimes very difficult to put numbers to the loss in performance after subsystem failure. For this reason a system using similar redundancy throughout with double failure survival capability is superior in that the performance is always very close to the desired value and the integrity study is

then a simpler task of fault analysis.

One basic problem that appears in multiple systems is that of 'nuisance' disconnects if the system becomes too complex and when not enough attention is given to component tolerances. A certain amount of relaxation from the optimum performance may well produce a valuable reduction in the probability of an incorrect failure indication (which must be considered as a genuine failure when integrity studies are performed).

The actual reliability targets now being aimed for (failure probabilities in the region of 10^{-8} per hour) are so severe that it is impossible to prove that such a figure has been reached to any reasonable level of confidence unless a statistical method is used in conjunction with redundant signal chains.

5.4 Serviceability

In order to achieve system separation in the current designs, the number of components and inter-system wiring has increased to an alarming level whereas in an integrated system, a single component or signal chain would suffice. In one design, the number of lateral stick position transducers has reached the grant total of 16 with a possibility of even more being added later. This large number is due to the design being based on a number of subsystems (in this case a mechanical signalling system, duplicate electric signalling system and a limited authority autostabiliser with a 'manoeuvre boost' system) whereas if a quadruplex manoeuvre demand system were employed only 4 or at the most 8 position transducers would be required. The servicing implications of this increase in numbers of components is obvious and the pre-flight check-outs are becoming difficult, making it difficult to attain quick turn-round times.

In general, the more the flight control system is integrated, the less difficult becomes the servicing problem and if similar redundancy is used to achieve the desired reliability, then inter-lane comparison can be used as a means of system check-out. A new approach to maintenance may be possible which may lead to a reduction in the amount of pre-flight testing required. If the errors between the signalling lanes are recorded for each flight in the form of percentage times of the total flight that the errors have exceeded certain values, then a histogram presentation will show if any lane has deteriorated to the extent that the probability of a fault warning will be greater than the desired value (or the order of 10^{-3} per hour in a quadruplex system). From the cumulative probability tables for normal distributions, 10^{-3} probability corresponds to a 3.3σ error, where σ is the standard deviation and if the σ, 2.6σ and 3σ error thresholds are chosen for the 'maintenance record', then 3.3σ could be chosen as the fault detection threshold. The σ and 2.6σ probabilities are approximately 3.3×10^{-1} and 10^{-2} and so should provide statistically significant information during a normal flight of 3 hours to predict with confidence the system behaviour on the following flight. With this maintenance recording procedure, the only checks before each flight that would be necessary would be the operation of the fault detection and correction logic. There would obviously still be a need to check absolute system performance on the ground periodically at 'first line' level at 200-500 hours interval to establish that no common deterioration had taken place.

In systems employing dissimilar redundancy in the form of a number of subsystems, all different, this maintenance recording technique would not always be possible and in fact this problem is reflected in the current search for the application of maintenance recording data because of the difficulty in current systems of finding suitable performance criteria.

6. FUTURE POSSIBILITIES

The final acceptance of electrical signalling with no mechanical reversion will lead to new developments. The following developments would be possible and the advantages gained would finally justify the development programme in electrical signalling.

(a) The overall aircraft design could be optimised taking into account the feedback control terms and lead to higher performance. Trim drag, drag due to excessive fin size, etc could be reduced at the expense of basic stability and control characteristics.

(b) Following the work described in Ref. 7, the structural weight may be reduced and the necessary structural mode stability could be achieved by the use of feedback terms and also the fatigue life could be improved.

(c) The cockpit layout could be considerably improved by the use of a miniature controller instead of a conventional stick. This controller also would allow more accurate flying in turbulence if the hand is properly supported as in that described in Ref. 6. The design of miniature controller is closely tied to the Manoeuvre Demand control law which determines the stability and response characteristics of the aircraft/control system combination.

(d) New multiloop control systems involving conventional elevators and ailerons, direct lift, jet reaction, etc can be fully exploited and the correct blending of these control outputs can be achieved together with the necessary feedback terms. The actual control surface and reaction control configuration can be optimised taking into account the automatic control requirements.

7. CONCLUSIONS

(a) More reliance is being placed on electronic control systems in current aircraft designs, but the full performance advantages of feedback control cannot be realised in most current designs since

the overall control system performance must not be degraded to an unacceptable level by the loss of one subsystem.

(b) Although systems are advocated with mechanical reversion for reasons of integrity, the problems of synchronising the mechanical and electrical systems can cause a significant increase in overall complexity and loss of integrity. Such systems are difficult to analyse from the safety point of view since the performance deterioration associated with a reversionary mode must be related to the failure probabilities and with current designs this process is becoming extremely complex and the chances of overlooking critical failure modes is becoming higher.

(c) The added complexity of subsystem separation is causing a large increase in maintenance requirements. In particular, pre-flight check-outs are becoming extremely involved.

(d) The use of integrated manoeuvre demand system using similar redundancy is a powerful way of overcoming many of the design problems in current systems such as are described above.

(e) Serious consideration should be given to the use of feedback control in the early design stages of aircraft in order to lead to an optimum flight control system design. A careful study of basic aerodynamic characteristics such as static margin and directional stability should take account of the current auto-control 'state-of-the-art'. Also, the control surface configuration and control powers should be related to the use of autocontrol systems.

(f) By seriously considering electrical signalling with no mechanical reversion in new project designs, it should be possible to optimise the overall aircraft design to give good performance and to use feedback techniques to satisfy the handling qualities and structural stability requirements.

(g) Work should continue on the two fronts:- to establish handling requirements of aircraft including autocontrol characteristics and to continue a programme to improve automatic control technology for aircraft application.

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ACKNOWLEDGEMENT

To B.A.C. (Preston Division) for helpful discussions in the preparation of this paper.

SIMPLEX AUTOSTABILISER (LIMITED AUTHORITY SERIES ACTUATORS) A/P PARALLEL ACTUATOR	SIMPLEX ELECTRIC SIGNALLING + AUTOSTABILISER A/P PARALLEL ACTUATOR	MULTIPLEX A/S OR CAS A/P FEEDS INTO "INNER LOOP" SYSTEM	MULTIPLEX ELECTRIC SIGNALLING LARGE AUTHORITY A/S OR MANOEUVRE DEMAND CONTROL	QUADRUPLEX ELECTRIC SIGNALLING MANOEUVRE DEMAND CONTROL INTEGRATED A/P SYSTEM.
MECHANICAL REVERSION AFTER 1st FAILURE	MECHANICAL REVERSION AFTER 1st FAILURE	MECHANICAL REVERSION AFTER 2nd FAILURE	MECHANICAL REVERSION AFTER 2nd FAILURE	NO MECHANICAL REVERSION
LIGHTNING, F. 4, etc.	MIRAGE III, etc.	F-III, etc	MIRAGE III V, CONCORDE, etc.	R&D PROGRAMMES IN U.K. R.A.E. HUNTER PROGRAMME U.S.A.F. "FLY-BY-WIRE" PROGRAMME AT F.D.L

Fig. 1 **Flight Control System Development**

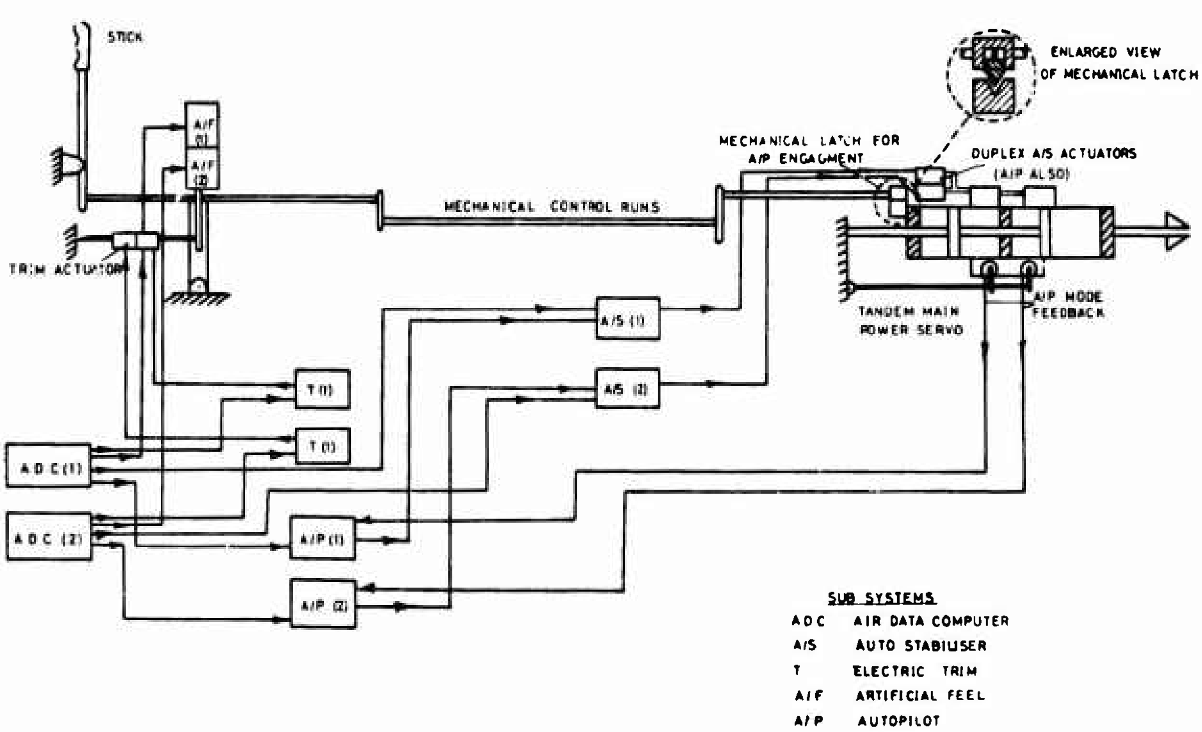

Fig. 2 **System Using Limited Authority Series Servos for Autostabilisation and Electrical Operation of Main P.F.C's for Autopilot.**

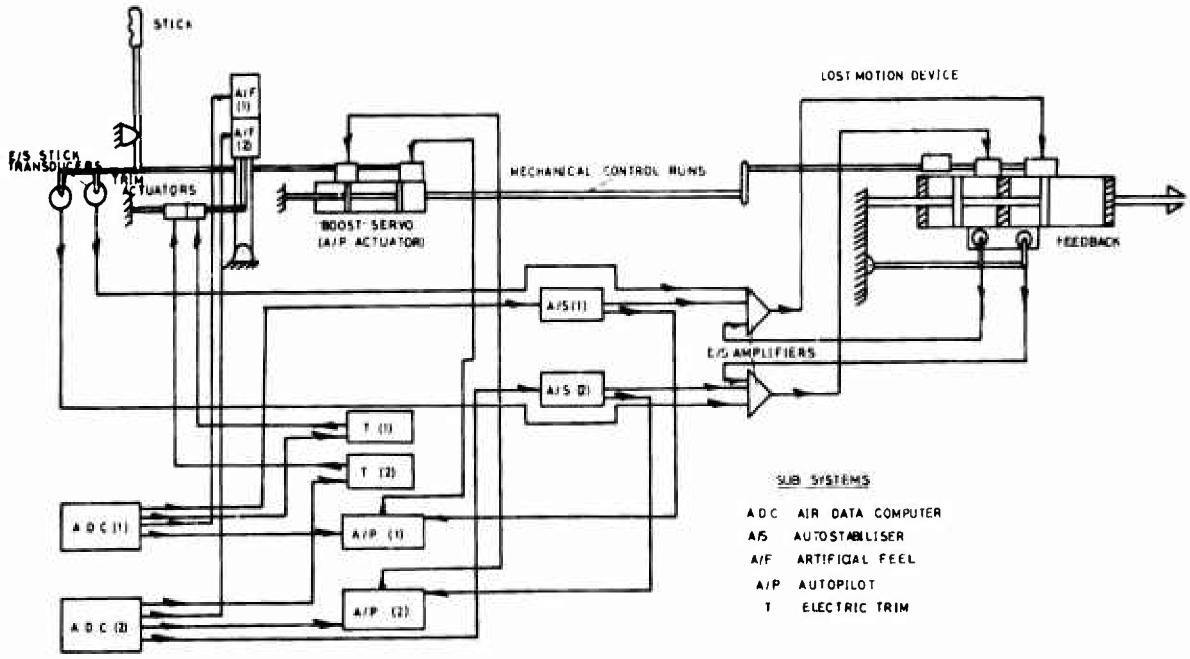

Fig. 3 System Using Electric Signalling of Main P.F.C's with Mechanical Standby.

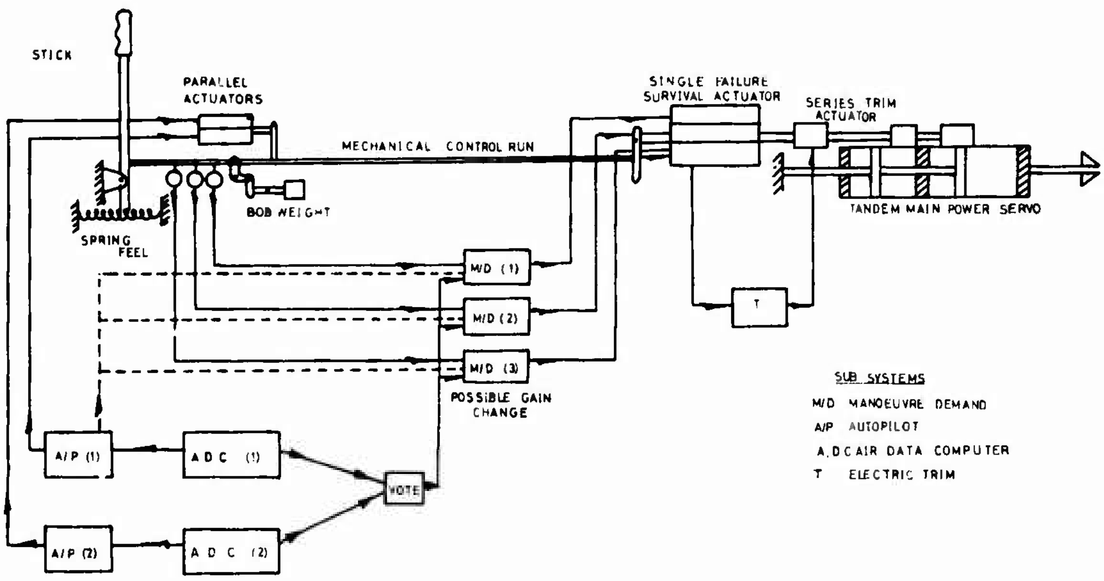

Fig. 4 System with Larger Authority Multiple Series Actuators giving "Command Augmentation".

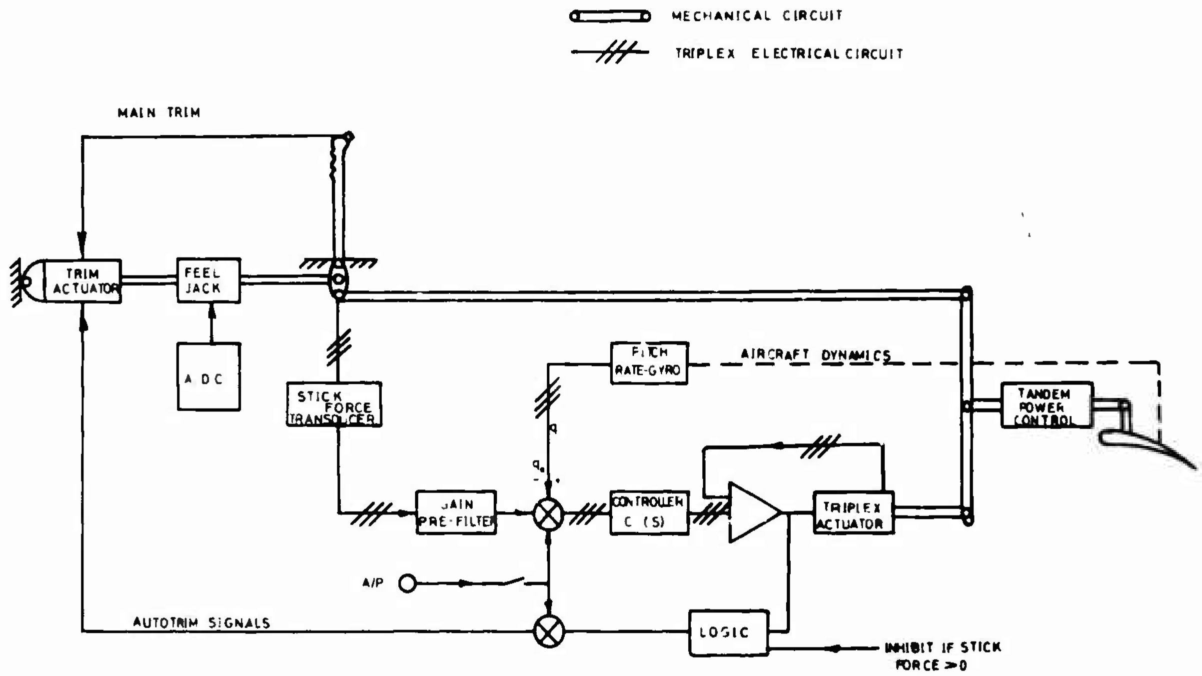

Fig. 5(a) Multiplex Command Augmentation System (Series Actuators) with Mechanical Reversion.

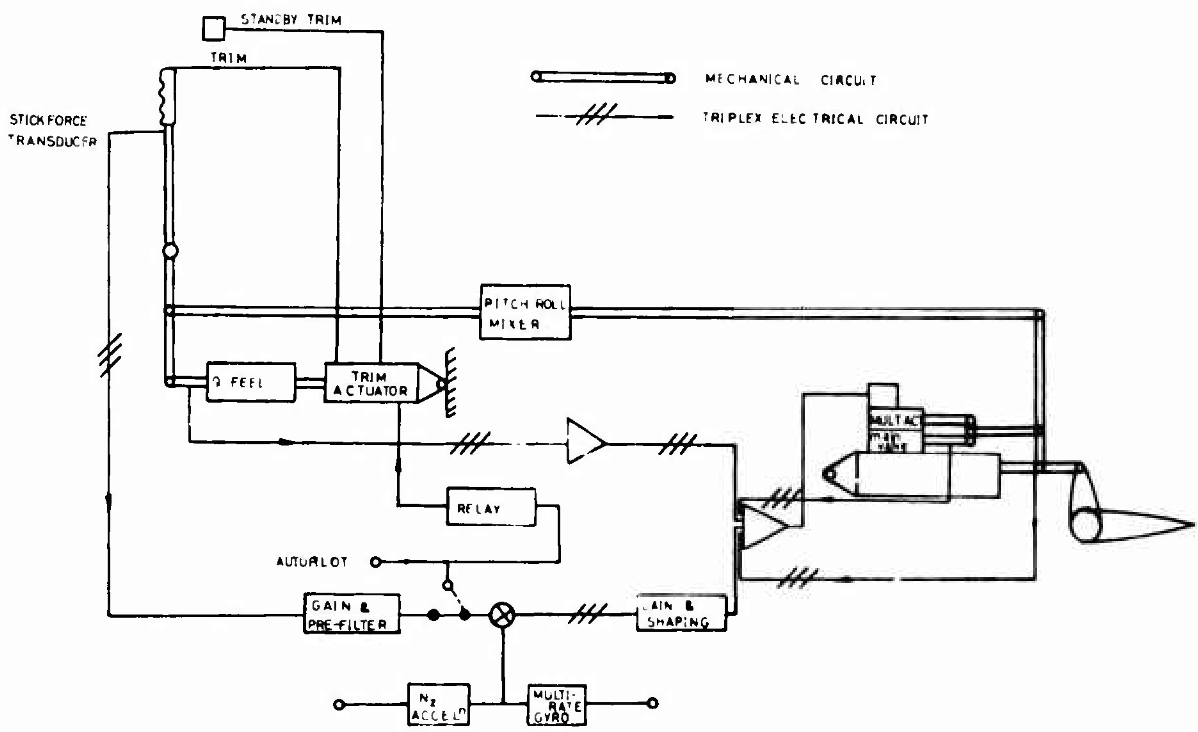

Fig. 5(b) **Multiplex Manoeuvre Demand System (Series Actuators with electrical feedback from main P.F.C's) with Mechanical Reversion.**

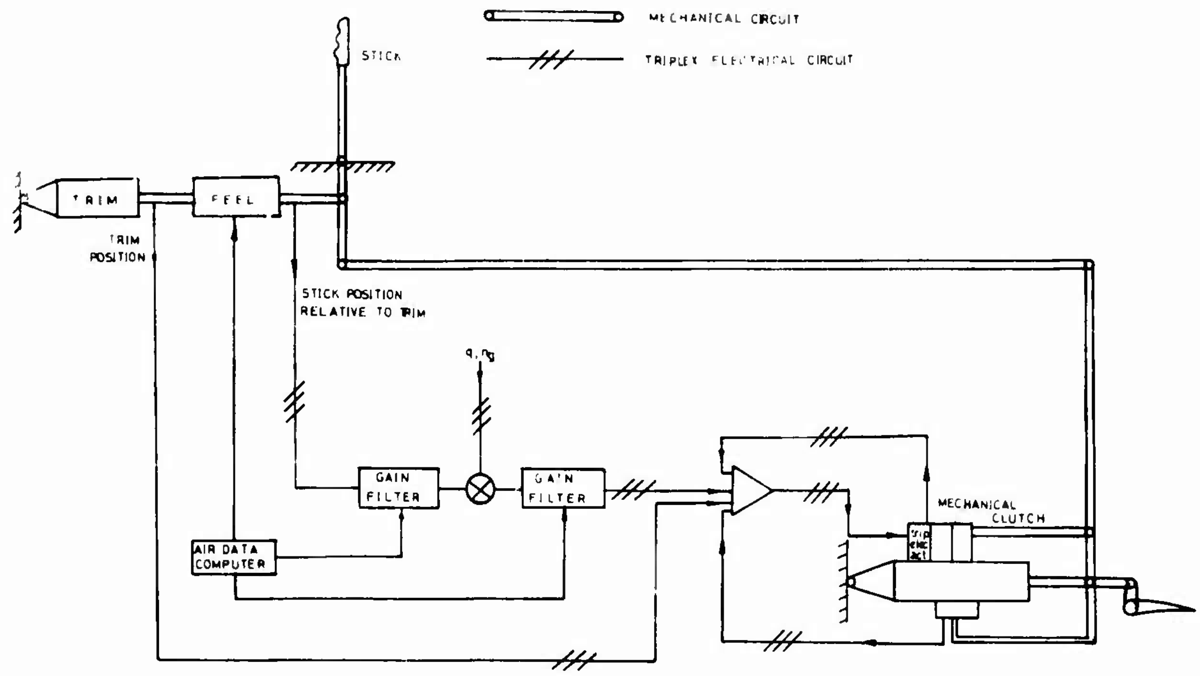

Fig. 5(c) **Multiplex Manoeuvre Demand System with Clutched Mechanical Reversion.**

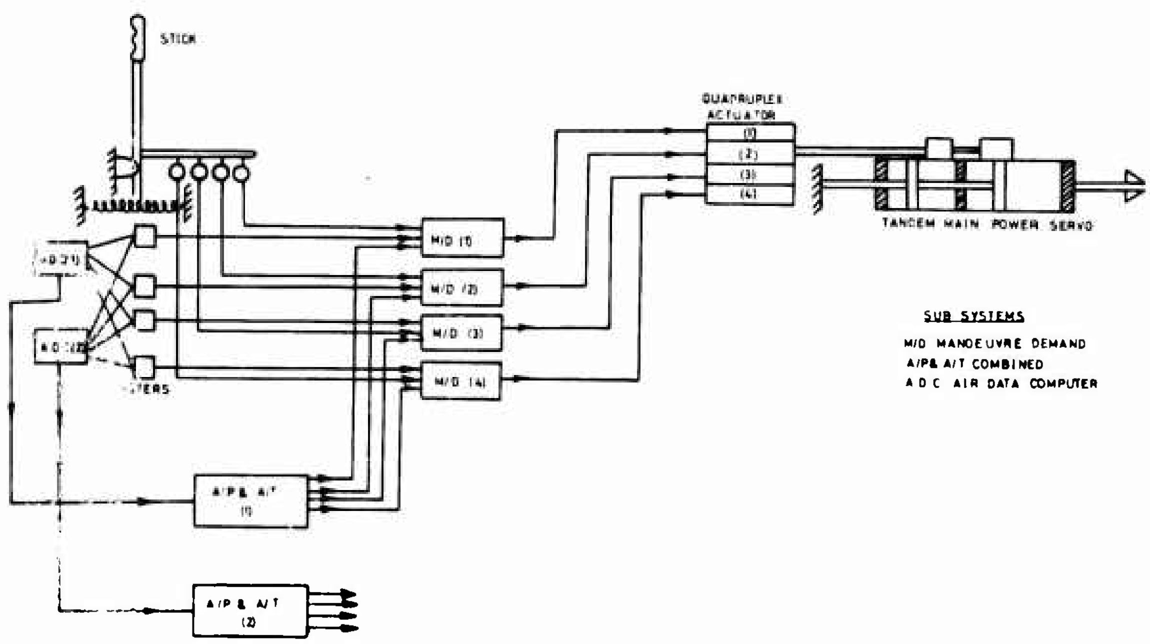

Fig. 6 **Integrated Manoeuvre Demand System - No Mechanical Reversion.**

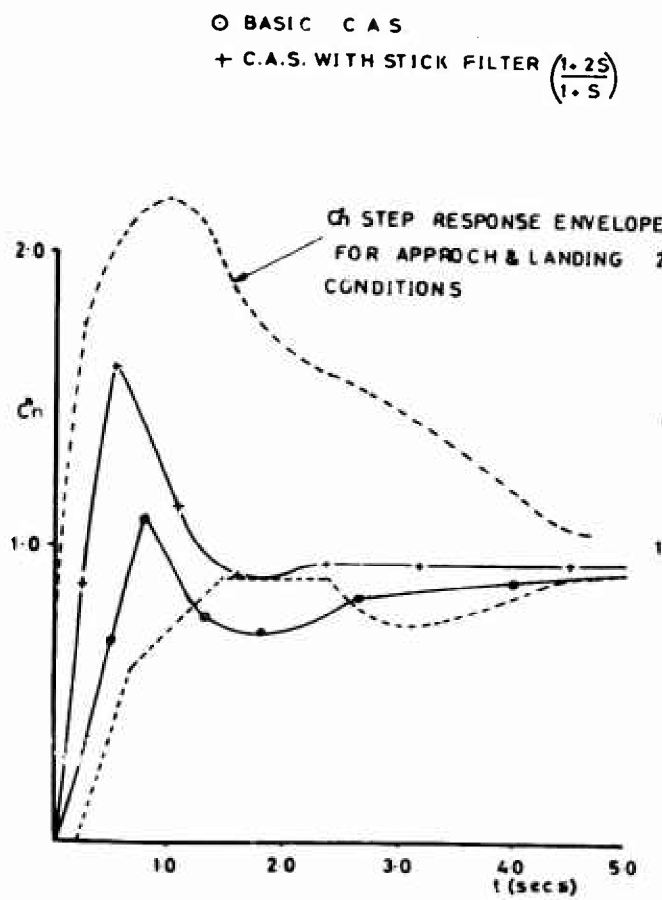

Fig. 7(a) C^*_N Response for M = 0.21 Sea Level

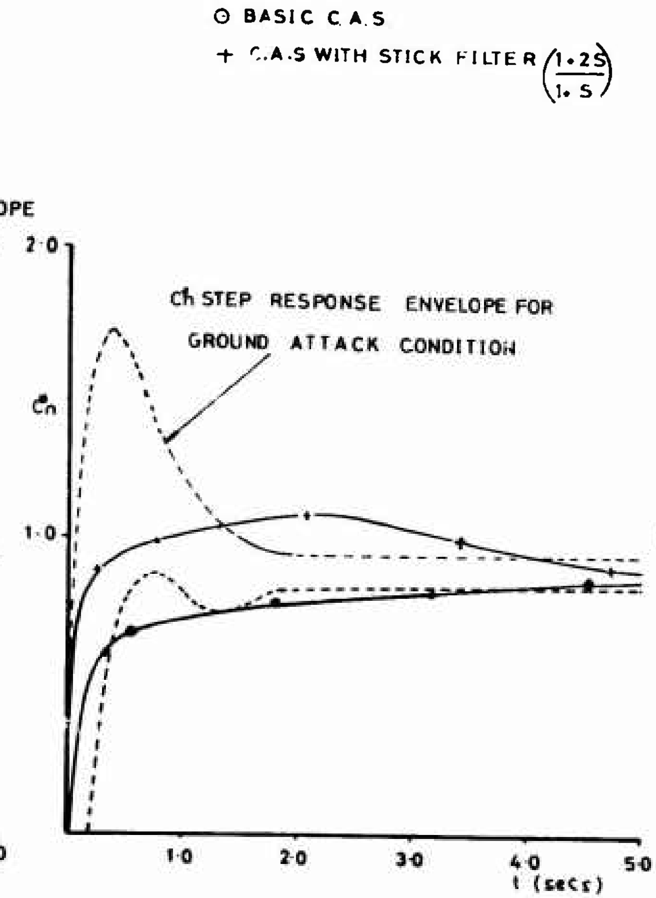

Fig. 7(b) C^*_N Response for M = 0.9 Sea Level

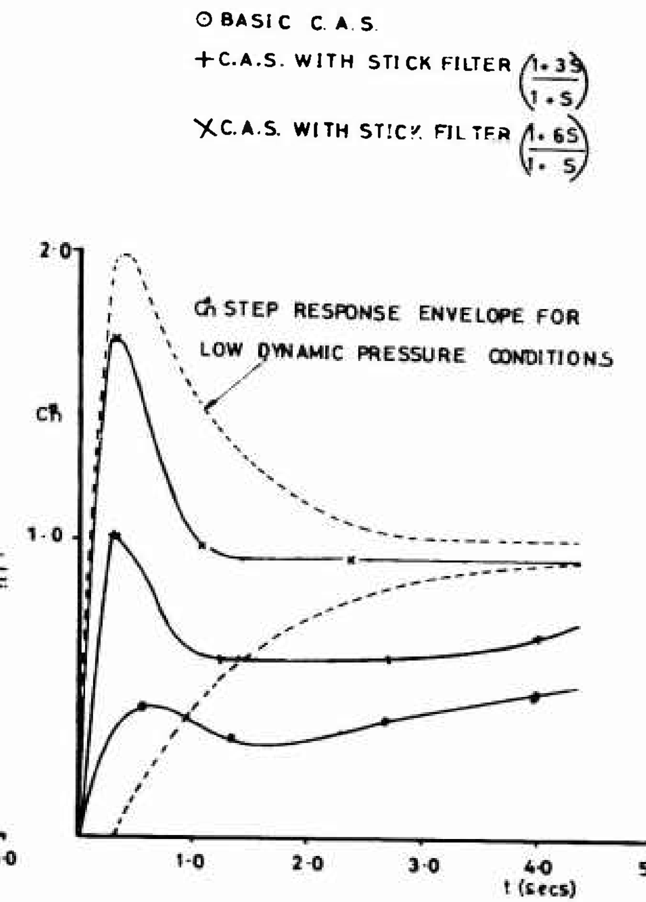

Fig. 7(c) C^*_N Response for M = 1.4 50,000 ft.

THE INTERFACE OF MAINTAINABILITY, RELIABILITY, AND ASSOCIATED DISCIPLINES IN AIRCRAFT DESIGN

by

S. LOVE, .ought Aeronautics Div. of LTV, U.S.A.

THE INTERFACE OF MAINTAINABILITY, RELIABILITY, AND ASSOCIATED DISCIPLINES IN AIRCRAFT DESIGN

The specification of the Model O2U, the original Corsair for the United States Navy, provides some interesting reading. I quote from the section on accessibility.

"All parts of the Corsair are designed and arranged to permit easy access for lubrication, adjustment or repair. Cowling throughout is provided with ample-sized access doors wherever needed and fuselage under pan is quickly removable for inspection and adjustment of controls, etc. With either side tank removed (which can be done in a few minutes), all parts within the fuselage cockpits are advantageously accessible. The walkways at root of lower wings make it possible to work on engine and various airplane parts in the environs of the cockpits while on the water or flying field. Access doors are provided in wing panels so that control arms, etc., may be easily lubricated and adjusted, and inspections made of the interior of wing panels."

That quote was generated in 1927. In 1964 design of the third Corsair, the Model A-7A, was initiated for the United States Navy. As part of our proposal, we submitted a movie covering the subject of maintainability. I quote from part of it.

"Let's discuss Availability philosophy in some detail. There are many definitions, publications, and opinions in existence on the interface of Availability, Reliability, and Maintainability. I would like to employ a general approach to Availability that should put us all on common ground.

"Airplane Availability is a function of Reliability and the number of items that have to be inspected, serviced, adjusted, or replaced. In a broad sense, this is the amount of maintenance an airplane will require. These two items are inversely related and, if we were able by some means and enough money to obtain 100 percent reliable components, then theoretically most of these maintenance items would not be required. Obviously, we are not going to get 100 percent Reliability, but we do have to make this factor as high as possible within the cost, schedule, and complexity constraints of this program to reduce maintenance to a minimum.

"These then become the key elements in Availability. We've concentrated on Reliability by trying to get better components, and even more important, by having fewer components in each system. We have made many improvements in the time element by concentrating on Accessibility, Turn-around, Workability, and Troubleshooting. Our success in this area produces an airplane that is designed to require a minimum amount of maintenance and special skills when it gets to the fleet."

These quotes represent a spread of forty years and there are significant similarities in the goals. As one indication of how the designer's problem has grown, let's compare three aircraft approximately 20 years apart in steps of evolution. They are the O2U (Figure 1), the F4U (Figure 2), and the A7 (Figure 3). They are all called Corsairs.

The statistics of these aircraft reveal some interesting data (Figure 4). The direct maintenance man hours/flight hour (DMMH/FH) have gone from approximately 1.5 to 9.5 with one additional difference. The 9.5 DMMH/FH on the Model A7 is a specific guarantee, like V_{max} and other performance guarantees. It is interesting to note in avionics not only that the weight has not gone up as fast as the number of components, but that the capability and flexibility of the later avionics has gone up geometrically. Note the significant increase in the cost of the avionics, both airborne and special support equipment. The number of parts has gone up ten-fold, and the availability has gone down, while maintenance requirements have gone up. Incidentally, the later Model A7 also has a guarantee on Mission Completion Success. The implication of these few facts on the system design problem today is obvious.

The aircraft designer today has indeed a <u>new</u> design problem. Weapon system complexity has increased almost geometrically in recent years as performance and mission requirements have become more demanding. It is almost axiomatic that these more complex systems <u>tend</u> to become less reliable, more difficult to maintain, more difficult to operate, and therefore less safe.

These undesirable tendencies can and, in fact, must be overcome during the initial design. The problems of attempting to modify state-of-the-art systems that have been designed without proper reliability, maintainability, human factor, and system safety considerations after the design is complete are almost insurmountable. For many years the industry has attempted to elevate Maintainability, Reliability, System Safety, and Human Factors to the same level as that of our other technical disciplines. It is mandatory to place all of these disciplines in the proper

perspective to maintain optimum balance and control. The work done to establish the requirement of these disciplines has been in part successful; but to the designer today, they must be categorized as a relatively new challenge. The complexity problem spawned the necessity to apply these new disciplines in the initial design phase. None of them are truly new, but they have become so much more involved with system complexity increase that they have taken on a whole new complexion. The actual development of the discipline into a useful engineering tool was and is a challenge.

How does one start with a grossly stated system operational problem and develop an engineering discipline compatible with the terms and parameters of the problem so that it is useable in initial design?

First, the problem must be understood. It must be possible to separate fundamentals from chaff. Understanding must be developed, in the main by deduction and history. The problem must be defined in engineering terms applying techniques which permit rational measurement and analysis. Such activity involves development of math models, testing, failure analysis, correlation with records, etc.

Once the problem is understood and described analytically, a prediction capability must be developed. No amount of analysis or test after the fact will change reliability, maintainability, operability, or safety one bit. If these factors are not considered initially, then they may well get out of control, and the design will never realize its full potential as an operationally deployed system. There analysis and prediction during design is required, much as is the case with Aerodynamics and Structures.

The next requirement is the formulation of these disciplines in such a manner that they are useful in the overall design process wherein many other factors clamor for consideration. This involves the development of a language with which to communicate with the other disciplines during the design. Other areas must be taught this language and, to a degree, educated in the discipline itself. These other areas must also understand the necessity for the design compromises which the discipline will tend to dictate. These other areas, especially design, must eventually desire to employ the discipline. The concept that a 2 percent increase in Mission Completion Success probability may be worth 10 knots in V_{max} is one that takes some education and some facts to back it up. Until such facts are understood and accepted by all others involved in the system design, proper design balance is not possible.

How does it work in practice? There are at least three major considerations that must be implemented. First, these disciplines must be integrated into the design operations as line functions with responsibility and accountability. Second, the organization of these line functions into the Engineering Department must be at a level consistent with their responsibility and accountability. They must report to and receive support from a management level consistent with their design responsibility. And, third, they must establish overall objectives and specific requirements.

Requirements are established generally by specification. To have maximum effectiveness, the requirements and programs resulting from them should contain both qualitative and quantitative aspects. Rational quantitative requirements should be specified wherever possible to provide a proper measurement for trade studies against which program requirements can be evaluated and/or demonstrated. In each of the four technical areas, a formal program plan is required. One then has a Human Factors Program Plan, a Maintainability Program Plan, a Reliability Program Plan, and a Systems Safety Engineering Program Plan. The plans, in each case, describe precisely the scope and content of the effort, the exact tasks to be accomplished, how each task will be accomplished, and provide a detailed schedule of major check-points and completion dates. In addition, the plans describe a method of testing and evaluating the results of the effort common to basic design. In addition to providing this "visibility", the written plan also provides management with the "what is being done, how it is being done, and when it is being done" information required to evaluate the quality and progress of the effort.

How does one accomplish the program? Or, perhaps, one should ask what are some of the techniques which the specialists in these areas use?

The goals and/or requirements affecting human factors, maintainability, reliability and safety engineering as stated in the detail specification for an aircraft system are usually very general. For example, "the MMH/FH must not exceed 9.5 and the mission completion reliability must be at least 75 percent." These values, while quantitative, do not provide the designer with the specific requirement which he needs in developing his design. At Vought Aeronautics we have developed mathematical models and computer programs to assist in the analysis of the subdivision of these general, overall requirements into allocations specific to a sub-system. Incidentally, math models and computer programs are meaningless without history or test data.

Using a maintainability simulation model (Figure 5), 0.28 MMH/FH was allocated to the hydraulic sub-system, and 0.41 MMH/FH to the electrical sub-system. These allocations are further detailed (Figure 6) and coupled with the associated maintenance concept, are then given to the sub-system designer as a target for his design--just as a weight target might be given to him. It should be made clear that these allocations are not assumed but are derived from an initial detailed analysis of the maintenance task for each sub-system. He now has a maintainability constraint which he must attempt to satisfy at the same time he is satisfying all others. In the same manner, reliability allocations are specified for a particular sub-system so that total reliability requirements of the weapon system can be achieved. General human factors, and safety requirements may also be made specific to a sub-system.

At Vought Aeronautics standard engineering procedures call for the provision of quantitative requirements to the designer through Design Definitions (DD). The designer responsible for a specific design originates the DD by listing all known requirements, applicable specifications, and general description of his system. All technical areas then provide their inputs in terms of requirements or constraints specific to their technical specialty. In this manner, the designer then has the first estimate of the requirements which his design must meet. The quantitative requirements specified by human factors, maintainability, reliability and safety may also be translated into meaningful qualitative requirements and included in the Design Definition. The basic principles must be educated into the designer, not restated to him every time. One is on the road to success when such basic principles are applied without prompting.

Following definition of the design requirements, the preliminary, basic design effort is started. Now we begin to have the day-to-day interaction of the designers and the specialists. Continuous design surveillance provides the specialists with the data needed to assess how their particular design requirement is progressing. It also provides the specialists and engineers in these areas the data necessary to point out good and bad points of the particular design at a design review meeting. In some cases these inputs can be evaluated and approved or rejected at the design review. In other cases the effect on cost, weight, structure, for example, may not be so obvious. In these later cases, a trade study may be required to evaluate thoroughly the effects of the conflicting requirements.

Participation in trade studies is an important technique by which these specialists may effect the selected design. Each concept is analyzed for the appropriate quantitative parameters, such as failure rates and effects, personnel skill levels and training requirements, and safety hazards. Ratings are then assigned in maintainability, reliability, etc., indicating the ranking of each concept. Depending on the specific trade study being accomplished, values are assigned to the particular rankings so that these disciplines are considered appropriately along with the other trade study criteria being evaluated.

These techniques which I have discussed- translating general requirements into specific, design definition inputs, design surveillance, and participation in design reviews and trade studies--are, of course, reiterative efforts continuing throughout the program.

Another technique which is used quite extensively is the use of design-tool mockups and simulators studies. Cockpit or crew station mockups and simulators have been used for some time. However, the extensive use of mockups of critical maintenance areas, such as avionics bays, or the use of simulated stores handling and loading tasks, is relatively recent. By employing these techniques both qualitative and quantitative data may be obtained early in the design phase. In test programs, such as these, maintenance time estimates and accessibility requirements may be checked. Human factors and safety consideration for display location and arrangement may be evaluated. Store loading, cargo handling, and turn-around times and procedures may be tested and revised until the most efficient concepts are identified. In many cases problems can be identified and corrections made with little or no wasted effort or changes in target dates of the total program schedule.

Today, significant components of weapon systems are subcontracted. To complement our "in-house" program and to assure meeting our contractual requirements, we also institute a similar program for our sub-contractors and vendors. Qualitative and quantitative requirements are specified in procurement specifications. Vendor proposals are reviewed, evaluated, and ranked according to how well they meet the human factors, maintainability, reliability, and safety requirements and programs specified. The follow-up through design analysis, surveillance and review of the subcontractor or vendor effort is addressed essentially the same as the in-house designer's efforts. In some cases, contract guarantees and demonstration plans are required and negotiated. One of the most important requirements is the demonstration of the reliability characteristic. The demonstration test is usually conducted at the subcontractor's plant using procedures and criteria established by specification. This type of testing is most often used to qualify equipment for production; but, it may be extended to monitor the reliability of equipment throughout the scheduled production.

If the designer and specialist have interfaced properly, the result is a practical, operable system which gives good service to the customer but, unfortunately, with higher initial costs and longer design programs. Not only is today's hardware generally more complex, but we have generated a more difficult design balance problem because of additional parameters. More data is generated because of the new factors to be reported and the engineering programs extend further into the system operational life. On one program today, we have 150 engineers in these four disciplines occupied full time--that's 300,000 man hours/year.

There has been little discussion thus far in this paper of special support equipment requirements. Today it may be axiomatic that as system complexities increase, so does the special support equipment in automation, complexity and cost. The subject of special support equipment and logistics is a subject unto itself; however, proper interfacing of these specialists and the support equipment designer is necessary for the generation of proper support equipment requirements.

The proof of the system design in these disciplines can only be verified with long term monitoring. This monitoring not only proves existing designs, but also builds more efficient tools for the future. In this regard the customer has new requirements imposed on him--namely, obtaining and reporting data over a long operational period. Contractors must have the benefit of operational experience supplied in data. Only the user can properly do this.

In summary, then, we are coming along with the proper interfacing of reliability, maintainability, human factors and system safety engineering into the designer's daily thinking. The design techniques are not yet highly efficient and still require development and refinement. Basic historical and operational data is still sparse, but state-of-the-art systems rarely do have much background data. Today we have guarantees in Maintainability and Reliability and formal requirements in System Safety Engineering. In the near future I predict we will have a new guarantee in System Safety; and reliability, maintainability and human factor requirements will become more stringent with attendant initial cost and schedule impact. In the next generation of avionics as a result of increased reliability, reduced special support equipment requirements at the field or intermediate level are predictable. Fortunately, the integration of these disciplines into the design process will improve with time and experience. Hopefully, the improvements in techniques will keep pace with the requirements and the designer will keep pace with the technical specialist.

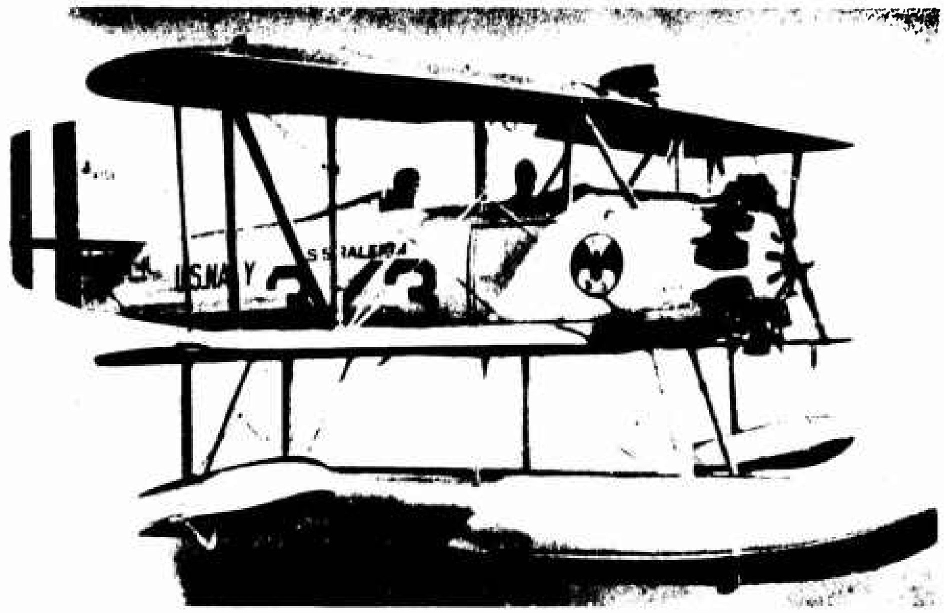

Fig. 1 O2U

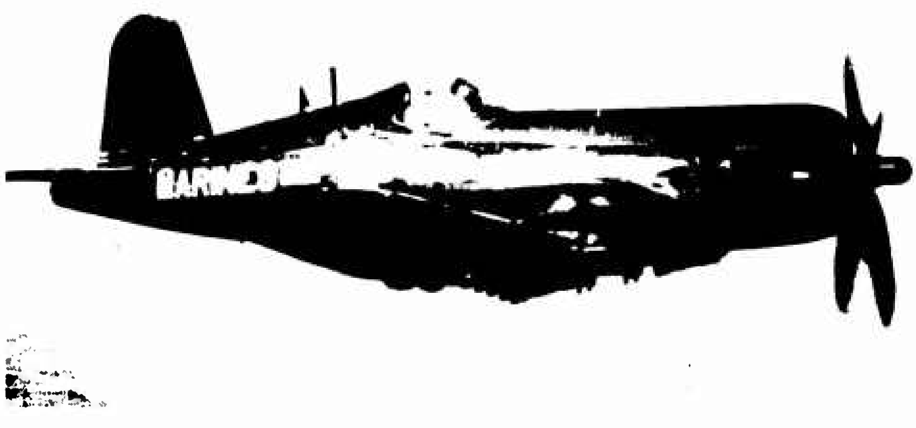

Fig. 2 F4U

Fig. 3 A-7A

	02U	F4U-1	A-7A/B	A-7D/E
• DIRECT MAINTENANCE MAN-HOURS/FLIGHT-HOUR	1.5	3.5	8.8	9.5
• ELECTRONICS				
COMPONENTS (BOXES)	NEGLIGIBLE	13	108	125
WEIGHT	NEGLIGIBLE	151 LB	826 LB	1,050 LB
PERCENT FLYAWAY COST	NEGLIGIBLE	3%	19%	28%
• SPECIAL SUPPORT EQUIPMENT PERCENT FLYAWAY COST	0	NEGLIGIBLE	8%	14%
• TOTAL PARTS	2,500	10,000	25,000	26,000
• AVAILABILITY	HIGH	80-85%	75%	75%

Fig. 4 Aircraft Complexity Growth

• SYSTEM	DMMH/FH
BASIC AIRCRAFT STRUCTURE AND EQUIPMENT	0.94
PERSONNEL SUPPORT AND EMERGENCY PROVISIONS	0.54
LANDING GEAR	0.72
FLIGHT CONTROL SYSTEM	0.63
PROPULSION	0.87
ELECTRICAL SYSTEM	0.41
HYDRAULIC SYSTEM	0.28
FUEL SYSTEM	0.34
COMMUNICATIONS SYSTEM	0.44
NAVIGATION SYSTEM	0.54
WEAPONS MANAGEMENT	3.79
• TOTAL	9.50

Fig. 5 Design Allocations of Direct Maintenance Manhours Per Flight Hour

• NOMENCLATURE		DMMH/FH	
• ELECTRICAL SYSTEM	0.41		
• ELECTRICAL POWER SUPPLY		0.27	
GENERATOR DRIVE			0.01
AC GENERATOR			0.11
DC POWER SUPPLY			0.03
DISTRIBUTION			0.06
EXTERNAL POWER SUPPLY			0.06
• LIGHTING		0.14	
EXTERNAL LIGHTS			0.06
INTERNAL LIGHTS			0.08
• HYDRAULIC (AND PNEUMATIC) SYSTEM	0.28		
• PC 1 POWER SUPPLY		0.11	
PUMP			0.03
DISTRIBUTION			0.04
EMERGENCY			0.04
• PC 2 POWER SUPPLY		0.15	
PUMP			0.08
DISTRIBUTION			0.07
• ACCUMULATOR PRECHARGE		0.01	
• PNEUMATIC SYSTEM		0.01	

Fig. 6 Design Allocations of Direct Maintenance Manhours Per Flight Hour (Electrical and Hydraulic Systems)

AVIONICS DESIGNER'S VIEW OF THE SYSTEMS INTERGRATION PROBLEM

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ABSTRACT

This paper delineates one example of the electronic designer's problem in providing an avionics subsystems design which can be suitably integrated into an aeronautical system to satisfy a critical operational requirement, i.e., low level, high speed penetration at a minimum altitude of 300 feet over or around any terrain profile.

It examines: (1) the required sensors and their capabilities; (2) considers the necessary interfaces for accommodation of the aircraft responses and control restraints; (3) describes the man-machine relationships and (4) delineates air safety required features. The basic sensor, a forward looking radar with a phased interferometer array and associated receiver is examined. Flight station display and instrumentation is described and the man-machine relationship considered.

Data of simulated low level flights over a selected terrain profile is shown under various conditions of environment (rain) and radar measurement error parameters.

ACKNOWLEDGMENT

The assistance of the following named people, in preparation of this paper is gratefully acknowledged.

Mr. Leo Botwin
Vice President of Engineering
Norden Division
United Aircraft Corporation

Mr. Lawrence Chanzit
Chief, Radar Research and Development
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United Aircraft Corporation

AVIONICS DESIGNER'S VIEW OF THE SYSTEMS INTEGRATION PROBLEM

By

Saul Weissman

Aircraft avionics have varied widely over the past thirty years. The original popular concept was design of the aircraft to perform specific flight maneuvers and attain prescribed altitudes, ranges, and speed without real regard for the ultimate use of the vehicle. After completion of aircraft design, consideration was given to installing the most readily available electronics black boxes in the aircraft. These would provide, on an independent basis, such capabilities as air to ground and air to air communication, low accuracy direction finding, and the like. Note the word is "install", an action which differs widely from what we now recognize as total aeronautical systems integration. Even with the later advent of more sophisticated electronics, such as the omni-range, radars, airborne computers, and self-adaptive flight control subsystems, installation on an individual component basis was still the rule. The attention to integration was limited to the particular interface that affected the individual black box operation.

It is only in recent years that the installer has been replaced by the systems engineer or avionics integrator. It has become increasingly evident that as aeronautical systems become more complex, and the operational demands on their performance become more critical, the avionics installation problem is only a small part of the total job.

The avionics designer must be continuously aware of a wide range of technical expertise, far beyond his basic avionics knowledge. This paper will examine one example of the application of such broad design considerations; the integration of a terrain following/terrain avoidance capability in the United States Air Force C-5A Transport System.

1. INTRODUCTION

Modern avionics systems are required to perform a wide variety of functions to satisfy the needs of new combat and/or non-combat types of aircraft. An example of the latter is the Air Force/Lockheed C-5A transport, in which the radar sensor and display system is required to provide the following modes of operation: ground mapping, weather detection and mapping, radar glide slope approach as a landing aid, computer aided release, terrain following, terrain avoidance, beacon, and stationkeeping display. A further stringent requirement is that several of these modes be simultaneously available to either the pilot, the co-pilot and the navigator station.

One of the important aspects of modern military aircraft operation is low altitude penetration, which has led to a new class of radar concerned with terrain following and terrain avoidance. Under tactical conditions, a low altitude approach to the terminal areas is required of the C-5A. These modes require maximum inter-disciplinary exchange between the avionics and airframe designers. This paper deals with this problem.

2. THE PROBLEM

As shown in Figure 1, each mode presents separate design problems even though they are integrated on a functional basis.

Terrain following provides automatic control in the vertical plane containing the aircraft velocity vector. In order to accomplish this, the radar sensor must provide the terrain contours in terms of elevation angle versus range coordinates. To derive the full benefit of low altitude penetration, maintenance of wings-level attitude at a pre-set altitude above the terrain is the basic design objective.

Terrain avoidance provides the capability for manually steering the aircraft around terrain obstacles. The sensor must provide three-dimensional terrain data in real time.

While this function can be provided by black boxes that can be readily installed, the quality of aircraft performance depends upon optimum interfacing between the black boxes and the aeronautical system taking into consideration the man-machine relationship.

The inter-disciplinary aspects of this optimization requirement involves the technologies listed in Figure 2. Some of the key technical elements and their relevant design parameters include:

a. Radar Systems Design - range, azimuth elevation and angle coverage, data rate, resolution, accuracy, and simultaneous modes of operation; weight, size, reliability, maintainability, prime input power, special field test equipment, cost.

b. Aircraft Installation - stabilization, scan volume, radome, thermal and vibration environment, cooling air, pressurization, boresighting.

c. Electromagnetic Scattering - Because we are "probing" and measuring terrain parameters by means of an electromagnetic pulse, the backscattering properties of terrain and discrete obstacles in clear weather and in precipitation must be known. Principal design parameter trade-offs include frequency and polarization of the radiated pulse.

d. Human Factors - optimum utilization of the pilot in the manual control loop, visual display of command and situation data, optimum transition between manual and automatic control, pilot fatigue relationship to acceleration environment.

e. Aircraft Flight Dynamics - Some of the key parameters are: dynamic response to control surface commands including time lags, maximum climb angle as a function of weight and tail winds, acceleration limits, speed regulation, and autopilot characteristics.

f. Control Theory - Due to aircraft saturation characteristics and acceleration limits, non-linear control theory must be applied and emphasis placed on sophisticated computer dynamic simulations.

The above considerations will become more evident as we go more deeply into the system.

3. OVERALL SYSTEM CONSIDERATIONS

Figure 3 shows how all the elements previously described fit together, particularly in the terrain following mode which provides automatic control of the aircraft. A functional description of the system operation is as follows:

a. Radar scans at 1 look/sec, providing full sector coverage.

b. Sensor obtains terrain data as elevation angle vs range at all azimuth angles, and the terrain data are transmitted to the TERRAIN THREAT EVALUATION and COMMAND COMPUTATION block. Data are in stabilized coordinates.

c. Angle and range to maximum terrain threat are determined, combined with angle of attack, and sent to the AUTOPILOT as the pitch angle command signal. Inputs include elevation information along ground track and set clearance altitude.

d. AUTOPILOT output drives aircraft ACTUATORS which change AIRCRAFT ATTITUDE, and thus change the AIRCRAFT/TERRAIN RELATIONSHIP to close the loop back through the RADAR SENSOR.

e. Terrain angle video, pitch command, and aircraft attitude are combined in a command/situation format on the DISPLAY.

The inter-relationship among the previously described technologies are clearly evident in Figure 3. It is also clear that avionics design integration must take place in the beginning rather than after airframe design is completed. Without early integration, we incur poor low altitude penetration profiles, which substantially increase the aircraft's vulnerability to the increasingly sophisticated ground defenses against aircraft. The cost per unit aircraft is high and going higher. The cost of providing an optimum design to achieve performance goals, rather than a design to unchangeable constraints, is merely that of proper and timely planning.

The basic interfaces with the multimode radar, as shown in Figure 4, include the inertial doppler navigation equipment, attitude and heading reference unit, angle of attack sensor, radar altimeter, stationkeeping equipment, MADAR (Malfunction Detection Analysis and Recording), autopilot, and flight instruments. Redundant inputs to the radar include: (1) two inputs of ground speed, one from the doppler equipment and one from the inertial system; (2) means of determining the aircraft flight vector, one using angle of attack and aircraft pitch and one from the inertial system; and (3) pitch and roll readings from either the inertial platform or the AHRU (Attitude Heading and Reference Unit).

It is evident that system optimization and integration must occur at all technical and management levels.

The degree of system optimization which can be achieved is dependent upon the manner in which this multitude of inputs are interfaced with the basic radar sensor and the weighting assigned to each element.

4. DESIGN CONSIDERATIONS

4.1 Phase Interferometer Technique

In the radar modes of operation that require vertical angle information, the phase interferometer technique is used. The phase interferometer was selected for the C-5A radar for several reasons. These include an extremely fast terrain data acquisition rate, wide elevation coverage, ability to ground map while terrain following, and excellent effectivity at shallow grazing angles.

The history of a single pulse transmission describes the basic operation of this method of angle measurement. Referring to Figure 5, the radar pulse is transmitted from an antenna whose elevation beamwidth is equal to or greater than the required total elevation angle coverage. As the pulse travels outward, it continuously illuminates the entire terrain profile with the elevation coverage of the transmitter antenna pattern. In Figure 5, (a) the position of the pulse is shown at four different times (or, equivalently, range) during its travel. That is, at range 1 the wavefront of the backscattered signal arrives at the interferometer antennas at a given angle with respect to the horizontal. At range 2 it arrives at another angle and so on.

The phase interferometer directly measures the angle of arrival of the return signal by measuring the differences in the time of arrival (or, equivalently, relative phase) of the signal at the two sections of the receiving antenna.

The received signals are separately processed through IF receiver channels and combined in a phase detector whose output is proportional to the phase difference between the two input signals. The result is an exact analog of the terrain profile as shown in the Figure 5 (b).

The shadow region exists in the terrain profile depicted in the figure. The processing in the receiver recognizes this situation and holds the output at the last good value.

4.2 Automatic Terrain Following Computer

The prime functional requirement of the Automatic Terrain Following Computer is to provide to the Autopilot an angle command signal (i.e., Pitch Command) that will enable the aircraft to clear the most threatening terrain along its path at the pre-set clearance altitude.

The information flow of the Functional Diagram of the ATFC shown in Figure 6, as follows:

a. Radar Sensor - measures the actual terrain profile in elevation angle vs range coordinates.

b. h/R Generator - h, the desired clearance altitude, is converted to its equivalent angle offset by the h/R Generator, where R is range. This is combined with the actual profile, the sum resulting in the offset Terrain Profile.

c. Terrain Weighting Profile - this unit generates a real-time waveform which is combined with the offset Terrain Profile to provide the maximum terrain threat data.

d. Climb Command Angle Computer - The output of this unit is the flight path command angle needed to clear the maximum terrain threat. This signal is sent to the Autopilot and the Display. It is computed on the basis of: pre-set acceleration limits, angle-of-attack, actual altitude (using radar altimeter), climb high factor in order to be wings-level over peaks, aircraft weight (as it affects the maximum aircraft climb angle capability). A Smooth Ride Control is provided which results in a peak-to-peak flight profile. Ground Track Notch is an angle gate which accepts radar data only along ground track.

e. Safety Features include:

(1) Altimeter Override

(2) Inadequate Clearance Warning (when obstacle is higher than maximum aircraft climb angle) - aural and annunicator indicators are used.

(3) Stall Warning Signal

(4) Equipment Failure Warning

4.3 Terrain Following/Terrain Avoidance Display

The C-5A Multi-Mode Radar has five basic types of displays, several of which can be used for a variety of modes while others apply to specific modes only.

The terrain following/terrain avoidance display, as presented in Figure 7, shows three types of information. At the top of the display are shown two terrain ridge lines with a zero degree elevation reference line. One ridge line shows the terrain elevation angle versus azimuth angle ahead of the aircraft out to 5 to 10 miles, depending on the range selected by the pilot. The other ridge line is for a range of 2.5 or 5 miles. The remainder of the scope shows a $\pm 45^{\circ}$ azimuth scan contour map with a 5 to 10 mile range and, in the center of the display, steering information.

In this manner, three dimensional terrain data namely, range, azimuth, and elevation contours is presented to the pilot in real-time. In addition, command data and aircraft altitude symbalogy are presented so that the pilot can make effective use of the various sources of data that are important to him. Note that the pitch command signal is obtained from the automatic terrain following control system and, as a result, the display serves as a monitor permitting effective transition between manual and automatic control.

This one display provides the pilot with all of the information required for control of the air vehicle to perform the terrain following, avoidance function.

5. TERRAIN FOLLOWING PERFORMANCE SIMULATIONS

As noted previously, the complexity of the various governing equations and their interaction, and the non-linear functional characteristics of the aircraft, requires sophisticated computer simulations in order to obtain design criteria and evaluate performance capabilities. The automatic terrain following mode of the C-5A has gone through such a process. Figure 8 shows the flight conditions that were used for the computed performance profiles shown in Figures 9 to 12.

a. Figure 9 - Two aircraft trajectories are shown, one is an Error Free Trajectory, that is, a perfect radar, which demonstrates the dynamic capabilities of the aircraft itself. In this case, it handles this mild peak very well. The other trajectory shows the effect of a 0.01 radian radar error. The deviation from the first trajectory is negligible.

b. Figure 10 - This figure shows the response of the system to a repetitive sequence of hills. Again, the error trajectory is negligibly different from the error free trajectory which shows the inherent acceleration limitations of the aircraft design itself.

c. Figures 11 and 12 - These figures are the same as Figures 9 and 10, respectively, except that the effect of a 12 millimeter per hour rain is shown in place of the random radar noise error. In both cases, there is negligible difference in the trajectories.

Figure 13 is a statistical summary of the trajectory data shown in Figures 9-12. This data quantitatively shows the negligible effects of random radar error and rain storms on the trajectory capability of the aircraft itself.

Approximately 20 hours of initial flight testing in a flying test bed have been completed. These are preliminary tests to determine the achieved operational parameters at altitudes in excess of those anticipated during operational use of the system. Initial plots indicate that although considerable logic and loop gain adjustments may be required, the basic system design will be compatible with C-5A flight dynamic constraints.

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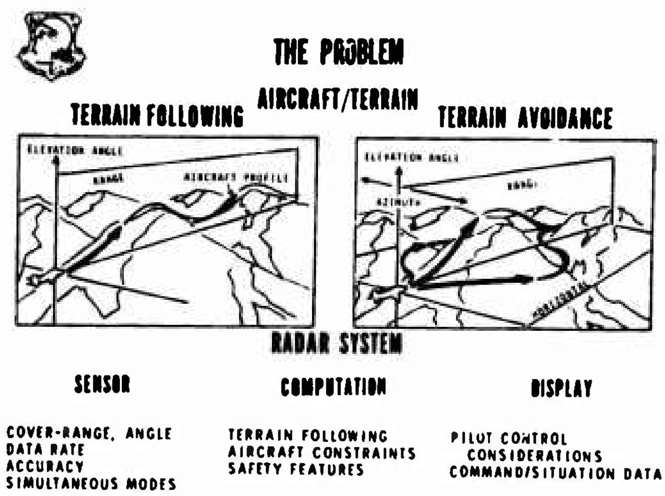

FIGURE 1

REQUIRED TECHNOLOGIES

1. RADAR SYSTEMS DESIGN AND AIRCRAFT INSTALLATION
2. ELECTROMAGNETIC SCATTERING-TERRAIN, STRUCTURES, ATMOSPHERE
3. HUMAN FACTORS
4. AIRCRAFT FLIGHT DYNAMICS
5. CONTROL THEORY - LINEAR AND NON-LINEAR

FIGURE 2

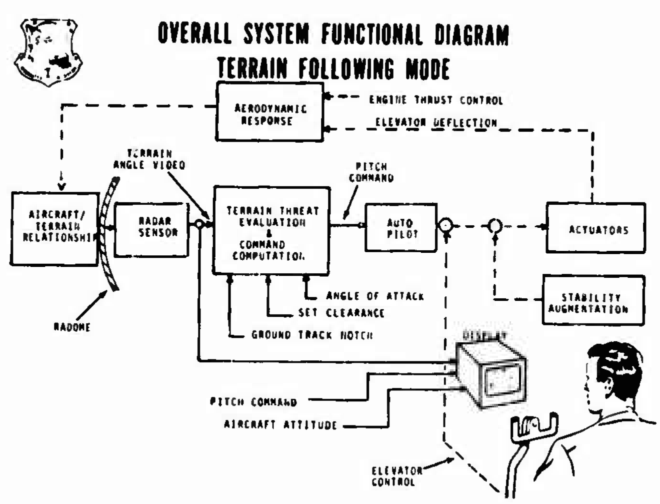

FIGURE 3

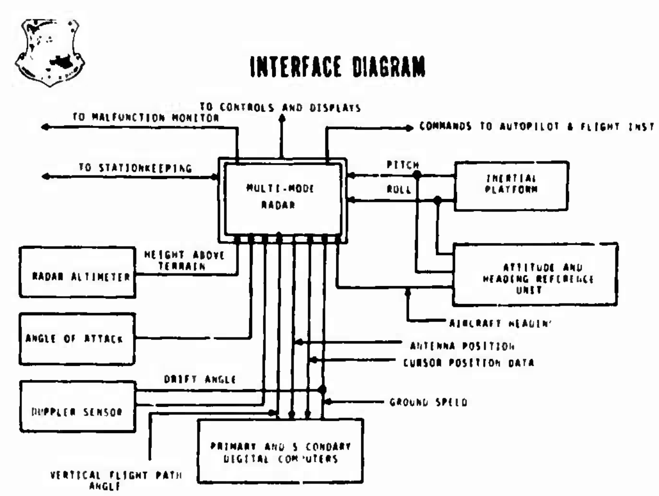

FIGURE 4

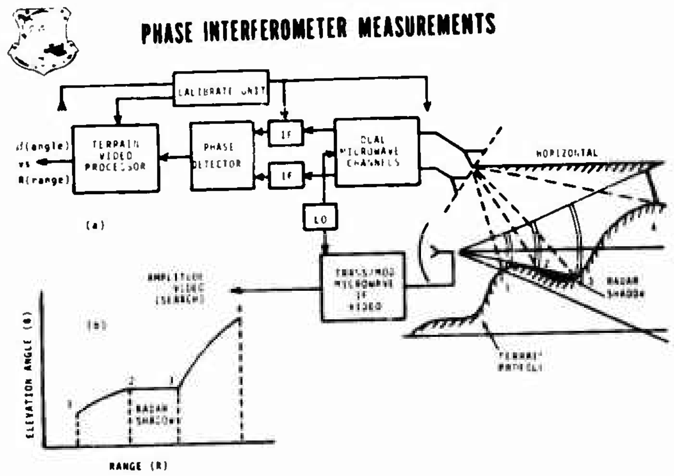

FIGURE 5

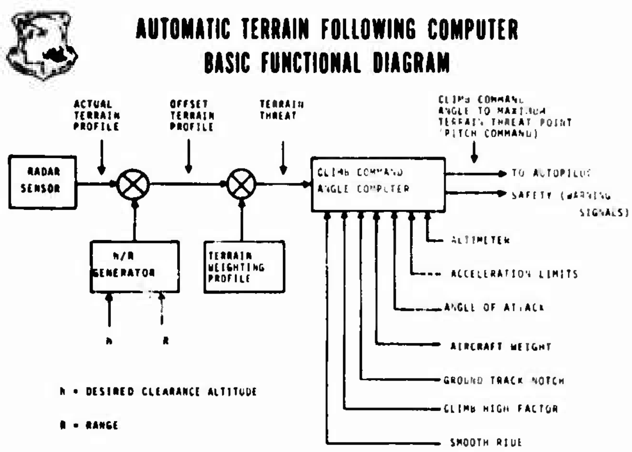

FIGURE 6

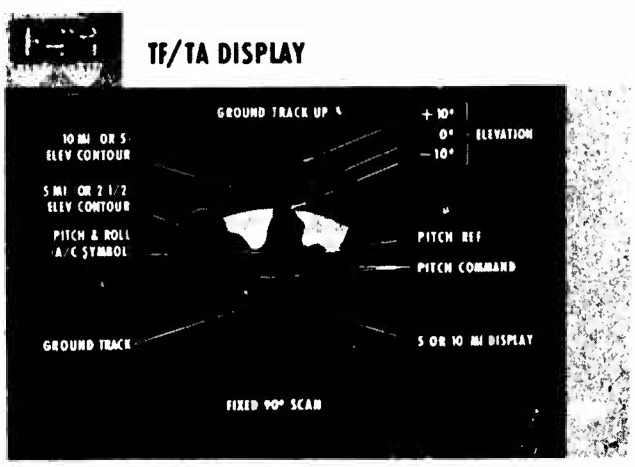

FIGURE 7

FLIGHT CONDITIONS FOR AUTOMATIC TERRAIN FOLLOWING SIMULATION

VELOCITY	475 FEET PER SECOND
WEIGHT	500,000 POUNDS
CENTER OF GRAVITY	0.25
MAXIMUM CLIMB ANGLE	+7.5°
MAXIMUM DIVE ANGLE	-7.5°
SET CLEARANCE ALTITUDE	300 FEET
TYPE OF TERRAIN	SOUTHWEST PENNSYLVANIA (CAL 6201)

FIGURE 8

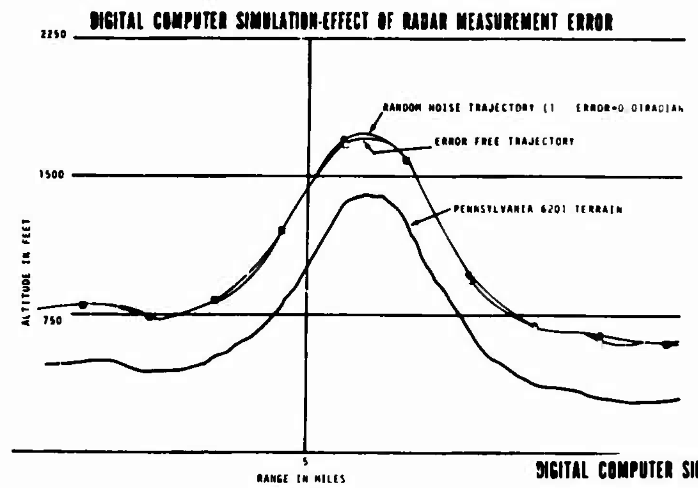

FIGURE 9

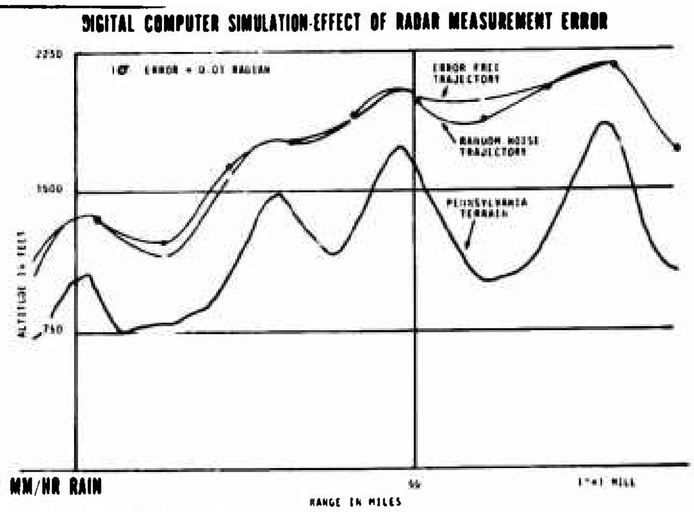

FIGURE 10

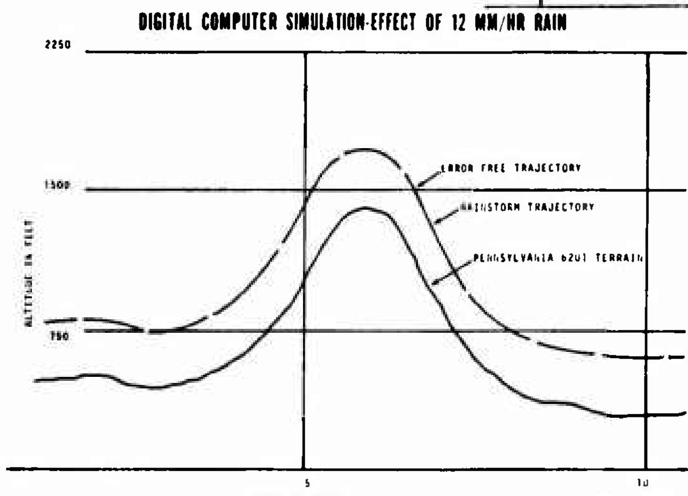

FIGURE 11

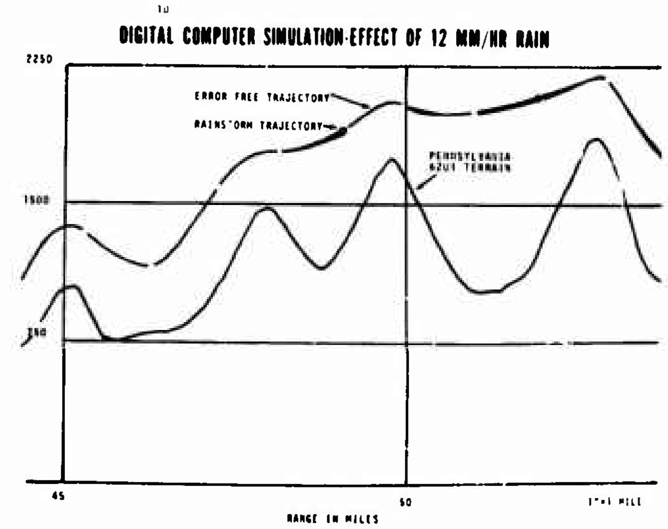

FIGURE 12

CLEARANCE STATISTICS FOR ATF SIMULATION RUNS

RUN	AVERAGE CLEARANCE	STANDARD DEVIATION	MINIMUM CLEARANCE
1. ERROR FREE (PERFECT RADAR)	385 FEET	132 FEET	289 FEET
2. RADAR ANGLE NOISE ERROR (0.5° (1σ))	395 FEET	124 FEET	272 FEET
3. HEAVY RAINSTORM (12 MILLIMETER PER HOUR)	386 FEET	133 FEET	289 FEET

<u>AIRCRAFT ACCELERATION DATA</u> (RUNS 1, 2, and 3)

RMS	0.12 "g" (APPROX.)
PEAK	0.6 "g" (APPROX.)

FIGURE 13

DEVELOPMENT PLANNING AND OPERATIONAL CAPABILITIES

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SUMMARY

The development planning process as a method for identifying, advocating, and obtaining approval for military system and equipment programs needed for new and improved operational capabilities is discussed. The experience of the U. S. Air Force in development planning is used as the example. The changes within the development planning process, the activities, and the tools used in the process are examined. A specific review of the "technology plot tool" is presented as a means of improving communications among the user (the operational command), the design engineer, and the planner. Development planning has evolved as the disciplined process which directs the talents and techniques of operational planners, system analysts, engineers, designers, and technologists toward the goal of providing effective and valid proposals for new systems and equipment for meeting the operational capability needs of the future.

Development Planning and Operational Capabilities

Major General F. M. Rogers

Development planning as a method and process for identifying, advocating, and obtaining approval for the development of new military systems and equipment has attained an important place in the decision-making process of nations seeking new and improved operational capabilities to meet their specific defense needs. Development planning as a simple function has been practiced in some form and degree since early times. However, the intervening years have witnessed a vast change in the procedures, the talents, and the activities employed in the accomplishment of development planning. The past decade has been a period of significant change in the environment in which new military system programs are advocated and approved for development. The development planning process as practiced by the U. S. Air Force has therefore undergone change. This paper will discuss the development planning experience of the U. S. Air Force to identify the changes in the decision-making process and what changes have been made within development planning.

Let us first look at the historical background of development planning so we may establish a perspective on the current need and character of development planning. In the late 1940's and in the 1950's the environment in which the Air Force intiated and developed new weapon systems was quite different from what it is today. The Air Force itself played a much larger part in the decision of what to develop and place in the operational inventory. The Air Force prepared "requirements" documents which stated the needed operational capability and stipulated the solution by describing the system which would provide such a capability. The planning for new systems included preliminary design, analysis, and the preparation of proposed program plans. These development plans were detailed documents which centered on how the development would be accomplished.

The decisions by the Office of the Secretary of Defense (OSD) were normally based on the data submitted by the Air Force with minimal detailed analyses conducted at the OSD level.

Other factors which characterized this earlier period from the present are as follows:

- The systems were less costly and therefore did not have as great an impact on the national budget.

- The systems were less complex from an engineering, operational, and maintenance standpoint; therefore, the near optimum system could be defined with relative ease.

- Technology was not as advanced and consequently did not offer as many different concepts or designs of solutions to do a particular job.

- The number of alternatives to be eliminated before arriving at the preferred design was not as great.

- The military environment and the threat which was postulated and/or validated as the capability of the possible adversaries were less complex.

The environment for initiation of new programs changed in 1961 by the adoption within the Department of Defense (DOD) of a planning, programming, and budgeting (PPB) system. This system brought together the missions and actions of all the military services under a single system where objectives and desires of all the services could be identified, approved or disapproved, and assigned funds for implementation. The Secretary of Defense also established technical and analytical groups within his own organization to monitor and implement the application of these new procedures.

Under this new planning, programming, and budgeting system, the Air Force requests to OSD for approval of new development programs were subjected to critical review and inquiries for more analytical information. The decision-making process at the OSD level had changed to a detailed analytical procedure which evaluated each aspect of a proposed system in depth. This change resulted in the need for the Air Force to provide data which were responsive to OSD and their requests for specific analyses. A new formal procedure for submitting requests for initiation of engineering development

of new systems was established by a DOD directive. The phase of effort for development planning was identified in this directive as "concept formulation" and preceded the other phases of a system life cycle such as contract definition, acquisition, and operational phases.

The changes in organization at the OSD level, adoption of the PPB system, and the DOD development initiation directive had an impact on policies, responsibilities, and procedures for obtaining new and improved operational capabilities and changed the philosophy from the old "requirements" document approach we mentioned earlier to a new approach which employs Required Operational Capability (ROC) and Required Action Directive (RAD) documents.

The development planning process is now characterized by many concurrent and iterative actions which are selected on a case-by-case basis and not by any simply structured procedure. The process may begin with the statement of an operational deficiency or need expressed by HQ USAF or by a major air command in a Required Operational Capability (ROC) document, orally or by letter. In response to a ROC, HQ USAF will normally issue a Requirements Action Directive (RAD) which has resulted from consideration of the views and comments by the other commands and organizations concerned with the ROC. The RAD may direct the Systems Command to perform planning actions such as advocacy of a new system and the preparation of a Concept Formulation Package/Technical Development Plan (CFP/TDP), preparing of a proposal for advanced development, accomplishment of a mission analysis, or accomplishment of a technology application study. These specific development planning activities are the heart of the development planning process and provide the basis for the proposals for new systems and equipment which will meet a deficiency or the need for a new or improved operational capability. The CFP/TDP document mentioned previously is the proposal document. This document is forwarded from the Systems Command through HQ USAF and the Secretary of the Air Force (SAF) to the Office of the Secretary of Defense (OSD) for decision. Within OSD the Office of the Director of Development, Research, and Engineering (ODDR&E) then prepares a Development Concept Paper (DCP) in conjunction with the offices of the Assistant Secretaries of Defense. The DCP is a summary document which describes the problem, recommended solution, and other facets of the proposal with any OSD recommendations for final review and decision by the Secretary of Defense. If the Secretary of Defense approves the proposal, a memorandum is issued to initiate contract definition or other appropriate action leading to hardware development. This approval is reflected as a change in the Five Year Defense Program (FYDP), the master document for recording the planning, programming, and budgeting decisions.

The foregoing brief summary of the process of development planning has centered on general procedures and does not bring out the details of the key activities that contribute to valid proposals for new systems and equipment. The key activities are those which were previously identified in the process as (1) advocacy and preparation of the CFP/TDP, (2) preparation of a proposal for advanced development, (3) a mission analysis, and (4) a technology application study. Let us look at each of these activities in greater detail:

Category "A" Advocacy

Proposals for new systems, subsystems, major modifications, or technical facilities are initiated by preparation of a CFP/TDP which is used as an advocacy document to present a convincing basis for obtaining program approval, entry in the FYDP, and release of appropriate directives and funds for contract definition or development.

Normally, the first step is to test the readiness of the effort to advocate a new development by preparing a clean, detailed presentation of the operational functions, concepts, and environment for the proposed new item. If this first step cannot be accomplished, the effort may be redirected to another planning activity which is a mission analysis. If the first step is satisfied, the next step is preliminary design which includes trade-off analyses and results in proposed designs, costs, and schedules and establishes what is feasible. Cost effectiveness models, based on the competing designs and operational concepts, are developed for use during preparation of the CFP/TDP and for use in the follow-on contract definition effort. Selection of

the representative preliminary design to be included in the CFP/TDP is accomplished by assessment of the utility of the competing preliminary designs and available systems and/or equipment. Management and procurement procedures which are most applicable to the specific program are also examined and identified.

The output of this activity is the CFP/TDP which we will now discuss in more detail.

Concept Formulation Package/Technical Development Plan (CFP/TDP)

The CFP/TDP is a versatile document which may vary in content but contains sections on rationale, system description, and program description. The rationale section explains why a particular design concept was selected and why it offers enough utility to justify expenditure of money for contract definition or development now and not at a later time. This rationale is based on objective analyses and persuasive advocacy based on a policy of open disclosure in which all evidence pertaining to the case is presented. Technology plots may be used to illustrate comparisons of competitive designs and the threat. The system description section describes the new item in terms of a representative design and includes the operational concept of the system.

The program description section has three subsections, the plan for implementation of the program, the costs, and the schedules. The plan for implementation of the program has its own unique characteristics. Emphasis is placed on how the next phase, contract definition or initial development, is to be conducted although the section includes information on the later phases of development and/or acquisition. Information on management approach, procurement plan, types of contracts, Request for Proposal (RFP) features, and source selection plan highlights may be presented in this section. The cost and schedule subsections are based on established estimating procedures and are as realistic as estimating procedures allow.

The CFP/TDP is a summary volume and therefore is not a very thick document. The content of each section is selected to present only the pertinent and significant information to the reader who is in the echelon of decision making. If required, backup information may be submitted in other volumes but not in the CFP/TDP. The CFP/TDP as a proposal document strives to be concise and objective without involving detailed and routine matters which may divert attention from the important issues for decision making. The sequential order of presenting rationale, system description, and program description represents the logic of the proposal. Any weakness in the rationale section regarding the validity of a threat and the selection of the optimum system to negate it may nullify the best system description. Also, a poor description may nullify the cost and schedule estimates in the program description if the system description did not identify a truly representative design. The system effectiveness and cost relationship of the proposed system versus alternate solutions to the problem must bear out a favorable advantage to the selected design.

By way of making sure that CFP/TDPs include data which may be of concern to the decision makers, informal contacts are made between Air Force and OSD for purposes of exchanging ideas on the particular area of concern. Items of concern which should be addressed are then included in the preparation of the CFP/TDP.

The CFP/TDP represents the Air Force's proposal for a system or a piece of equipment which will negate a threat or eliminate an operational deficiency. The support and talent expended in preparing the CFP/TDP therefore includes the best available in government and industry and is a product of a truly national effort.

Category "B" Demonstration of Technical Feasibility

This planning activity advocates the demonstration of the technical feasibility of some technique or concept as a necessary step before proposing a new system or item of equipment as a Category "A" advocacy effort for the operational inventory. The output of a Category "B" effort is a document, Proposal for Advanced Development. This document is prepared by the Systems Command and contains the following types of information:

- Description of the concept or technique for which the demonstration of technical feasibility is intended.

- Description of the technical approach and the proposed demonstration.

- The cost and schedule estimates for the demonstration.

- Rationale on the potential payoff if feasibility is proven and why a particular approach was selected.

Successful advocacy of a demonstration is followed by an advanced development program which accomplishes the demonstration. The results of the demonstration then become a technology input to a Category "A" Advocacy effort which may propose the development of a new system or item of equipment for the operational inventory.

Category "C" Mission Analysis

Mission Analyses examine in depth a particular operational mission or function to identify new and promising concepts to correct existing or potential operational deficiencies. The results of a mission analysis may warrant initiation of other planning activities such as advocacy and proposal of a new system as described under Category "A" planning activity or Proposal for Advanced Development and demonstration as described under Category "B" planning activity. The quality and validity of these other planning activities are often dependent on how well the mission analysis was performed and whether truly effective concepts of systems were identified.

There is no one procedure for accomplishing mission analyses. They may be accomplished by personnel from any level of Systems Command, other major commands, Air Staff, other services, or a task force comprised of qualified persons from several services inside and outside of government. In every case the using command is intimately a part of the mission analysis team, and in many instances the using command furnishes the mission analysis director. There is also no standardized format for the resulting mission analysis report since each report is tailored to best present the activities and results of a specific mission analysis effort.

In summary, the mission analysis is truly a "key" activity since it provides the valid basis for Category "A" advocacy efforts for new systems, Category "B" proposals for advanced development, and guidance to the exploratory development by recognition of far-out concepts which demand new technology.

Category "D" Technology Application Studies

The technology application studies are accomplished to exploit new technological breakthroughs and improvements for military application. Such a study examines a specific technological advancement to determine potential applications to various operational Air Force missions and functions or even completely new military concepts depending on the uniqueness of the breakthrough. These advancements do not happen in profusion, but nuclear energy, satellites, lasers, and high strength boron fibers are examples of noteworthy technical advancements which have opened doors to new or potential military capabilities.

Technology application studies often result in other planning activities such as Category "A" Advocacy for a New System and Category "B" Proposal for an Advanced Development Program to demonstrate the feasibility of a new technique.

The planning process and activities (Category A, B, C, and D) as described have been in practice for the past three years in the Air Force; programs for bombers, fighters, missiles, command and control systems have been or are being proposed by use of this process.

The foregoing paragraphs have examined the broader aspects of development planning activities. The following paragraph will review a specific tool and methodology which addresses the problem of defining the characteristics of a proposed weapon system during the preparation of the rationale section of the Concept Formulation Package/Technical Development Plan (CFP/TDP) for the Category A activity and provides a means for good communication among the development planners, the future users of the proposed weapon system, and the decision makers.

This problem is characterized by the user not knowing what technology can provide; consequently, he does not know what specific performance to ask for. On the other hand, the designer does not know what the user needs and therefore does not know what performance to provide. This communication problem must be solved if the planner is to successfully accomplish the establishment of the characteristics of the proposed system. The planner must also assure that he has the design solution which best satisfies the user's need. If systems design were an exact science, then this problem would be solved along with the communications problems. However, systems design is an art and the planner will have to evaluate more than one design approach to identify the best solution. His problem is to find an effective method for doing this.

There is a simple and direct approach which resolves these problems. The approach is based upon the use of a powerful tool called the "technology plot."

Technology plots are graphs which display the interrelationships among the critical operational parameters of interest to the user. Every point on a technology plot represents an aircraft design which the user could select as the design solution. For example, technology plots for an attack transport would display the trade-offs among range, runway length, payload, and flyaway cost. A sufficient number of technology plots are required to cover the full spectrum of potential technical approaches as defined by configuration, engine cycle, number of engines, etc.

The value of technology plots is threefold. First and most importantly, they close the communications gap because they give the user, planner, and designer a common basis for discussing the problem. Second, they provide a means for comparing the various technical approaches. For example, diverse technical approaches can be compared by simply overlaying the technology plots and examining differences in the trend lines. It is also possible to use the technology plots to determine the optimal design solution for <u>each</u> technical approach and then to compare the optimal designs; this is an important characteristic of the technology plot. Sufficient data is available with which the user or planner can select the design based upon a rationale of his own choosing, rather than having to accept a design which was selected as optimal based upon the rationale of the designer. Third, technology plots provide a convenient and complete data base for conducting cost and effectiveness analysis aimed at defining the most efficient design solution.

In order to employ technology plots in concept formulation, the designer must generate design and performance data for a full spectrum of potential design solutions. The process is, of course, a time consuming one requiring iteration and refinement; however, the generation of a large amount of relatively hard data is feasible with digital computers and comprehensive system design programs.

The next step in preparing the technology plot is for the designer, with the help of the planner, to identify the key variables and develop the most illuminating technology plots. This is accomplished through a trial and error process. The following ground rules are used in preparing the plots. First, as many key variables as practical are shown on one chart. Second, other plots and cross plots are used where necessary. Third, sub-optimization should be kept to a minimum and occur at the last possible moment in order to minimize possible distortion of the trend lines. Finally, a complete explanation should accompany each technology plot describing the basis for the computation necessary to construct the plot.

The following short example illustrates the approach as applied to a proposed fighter aircraft which is to have better maneuverability than advanced threat aircraft and be capable of gaining and maintaining this performance over enemy territory. Consequently, the trade-off between mission radius and maneuverability must be examined to determine what potential performance the state-of-the-art can provide in the time period of interest. This is accomplished for the proposed aircraft by constructing a technology plot as shown in Figure 1 for a specific gross weight.

This plot is constructed with maximum load factor in "G's" on the ordinate. Load factor is computed for a critical design condition and is indicative of instantaneous

turning capability. Mission radius based upon a design mission profile is shown on the abscissa. The curves are constructed by connecting designs having constant thrust-to-weight (T/W) ratio and constant wing loading (W/S). Every point upon this plot represents a potential design solution. Notice that the plot shows we can obtain a high instantaneous turn capability and a good mission radius with an aircraft having a low wing loading and low thrust-to-weight ratio; but good maneuverability also implies good rate-of-climb performance and good acceleration. These performance parameters can be shown on the plot, Figure 2, by connecting designs (dash lines) having constant values such as Performance Parameter A. Parameter A gives a direct indication of instantaneous rate of climb at a critical low "G" condition. Notice that to improve instantaneous rate of climb we would reduce mission radius and instantaneous turning "G." Performance Parameter "B" shows rate of climb at a critical high "G" condition. Notice that for a given weight aircraft, if we desire to increase instantaneous rate of climb for this condition, mission radius must be reduced as before, but instantaneous turning "G" must be increased. Considering both the high "G" and low "G" performance conditions, it is clear that an increase in maneuverability can only be achieved at the expense of mission radius.

Figure 3 is the same as Figure 2 except that the isolines showing thrust-to-weight ratio and wing loading have been removed from the plot; this simplifies the graph. Although these parameters are important to the aeronautical engineer, they may only confuse the user in interpreting the trade-offs. Moreover, we've found that our prejudices concerning the design characteristics necessary for a given capability tend to lead us to overlook potential design solutions.

Many technology plots can be used to compare different design approaches. For example, in Figure 4 we have the operational trade-offs for a different proposed configuration. By comparing this plot (dash lines) with other technology plots such as the previous one (solid lines), we can determine which particular approach provides the best performance. In this example, for a given "G" capability and radius, the configuration shown in solid lines has better maneuverability than does the configuration represented by the dash lines. This can be seen by noting that performance parameter isolines shown in solid lines lie to the right of those shown in dash lines.

The region of interest can be narrowed by applying additional constraints to the technology plot. This is done by defining the known limits of the performance parameters of the threat aircraft. By masking the region where the proposed system would not have superior maximum "G" turn performance (Figure 5, region below line X), low "G" rate of climb (Figure 5, region above line Y), and high "G" rate of climb (Figure 5, region below line Z), the region containing the potential design solution for the proposed superior aircraft is defined by the clear area.

There are other considerations such as cost and effectiveness which have not been illustrated in this short discussion of technology plots. However, technology plots provide an excellent point of departure for such considerations.

Technology plots aid the development planner in arriving at the design solution which is consistent with the state-of-the-art, and which best meets the user's needs. Technology plots do this by displaying a range of potential aircraft having performance capabilities which are achievable. They also provide the user a graphic description of the advantages and disadvantages relative to changes in characteristics he may wish to make after considering his needs in light of what technology can provide. Moreover, technology plots provide a complete data base for subsequent mission and force structure studies where costs and effectiveness can be thoroughly investigated.

Closing Statement

Development planning has evolved within a changing environment as an effective process for identifying and advocating the development of new military systems and equipment. The activities included in the pursuit of development planning utilize the talents and techniques of the operational planner, system analysts, systems designers, technologists, logisticians, management specialists, and procurement experts within a disciplined process directed to the goal of providing needed new or improved operational capabilities. Procedures and means of communication are constantly being improved to permit the best understanding between the future users of the proposed systems and the design engineers who establish the characteristics of the new system based on the available and anticipated technology.

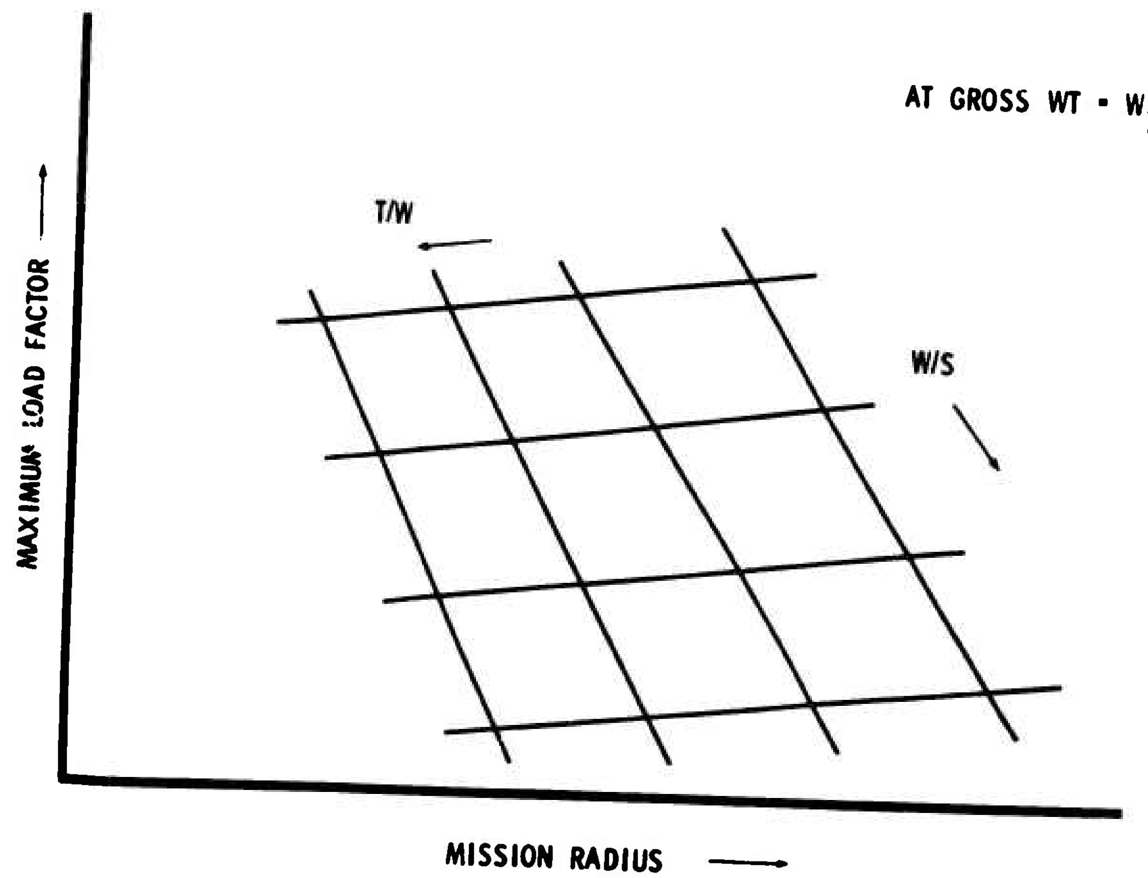

Fig.1 Technology plot

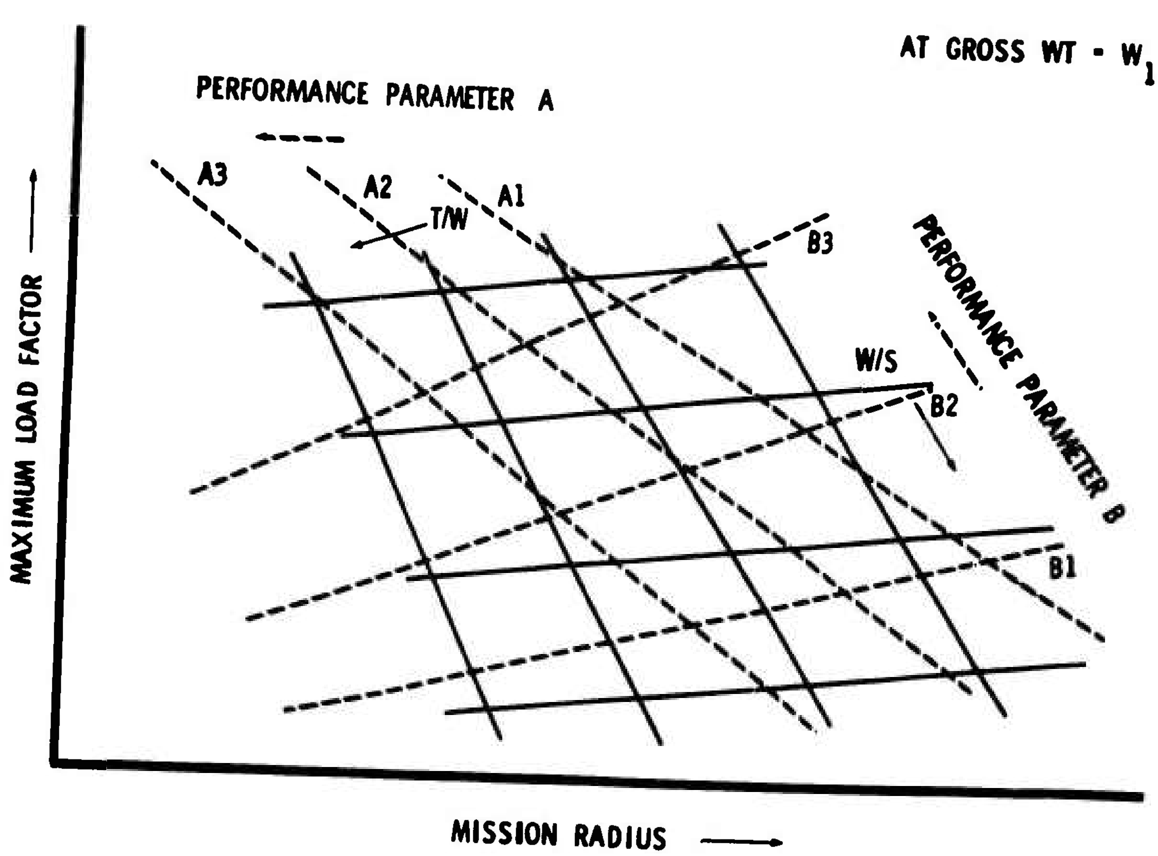

Fig.2 Technology plot

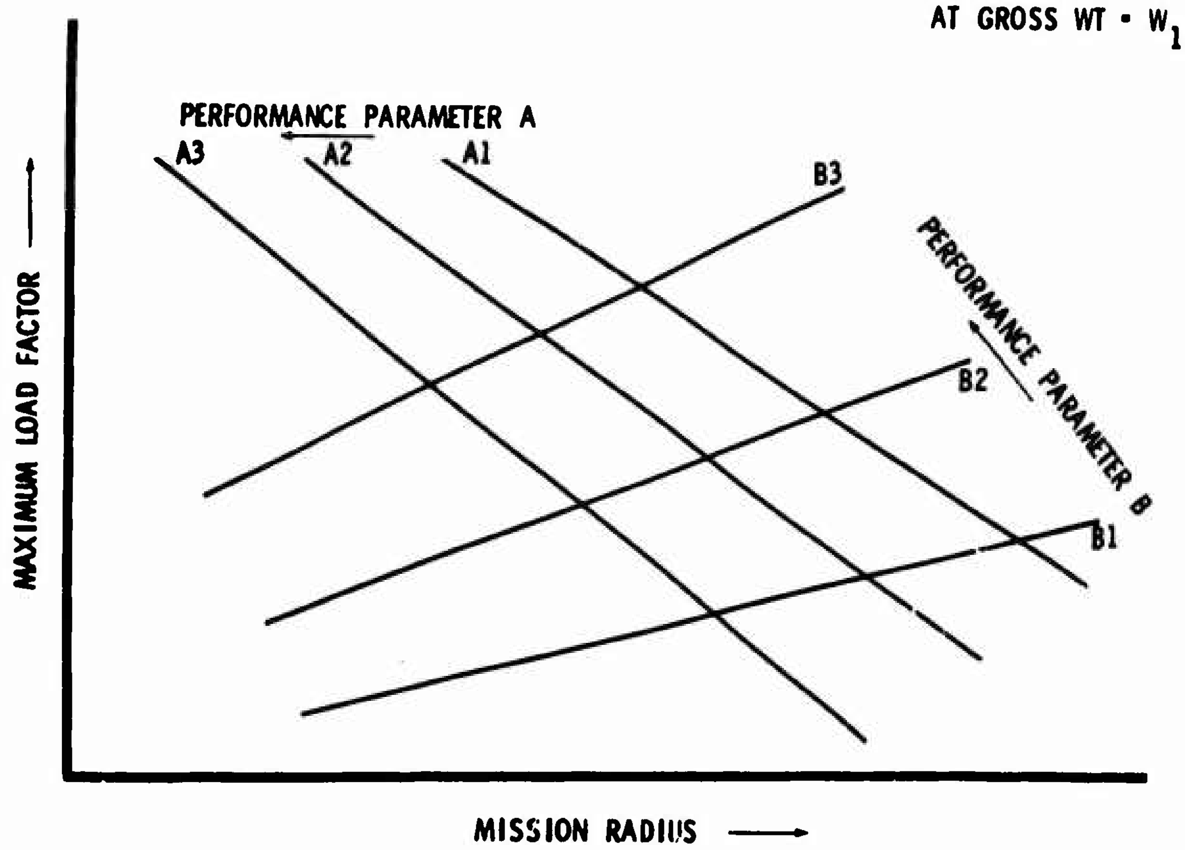

Fig.3 Technology plot

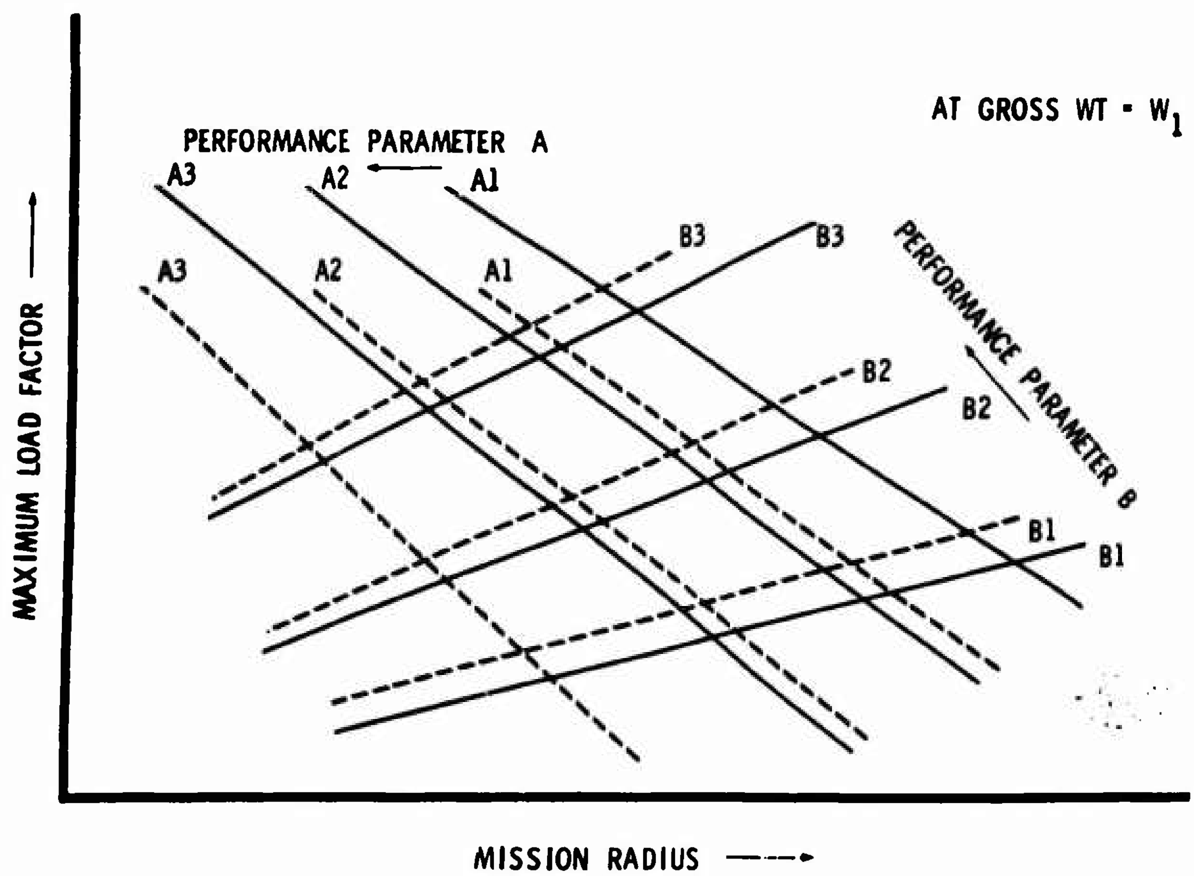

Fig.4 Technology plot - comparison of two configurations

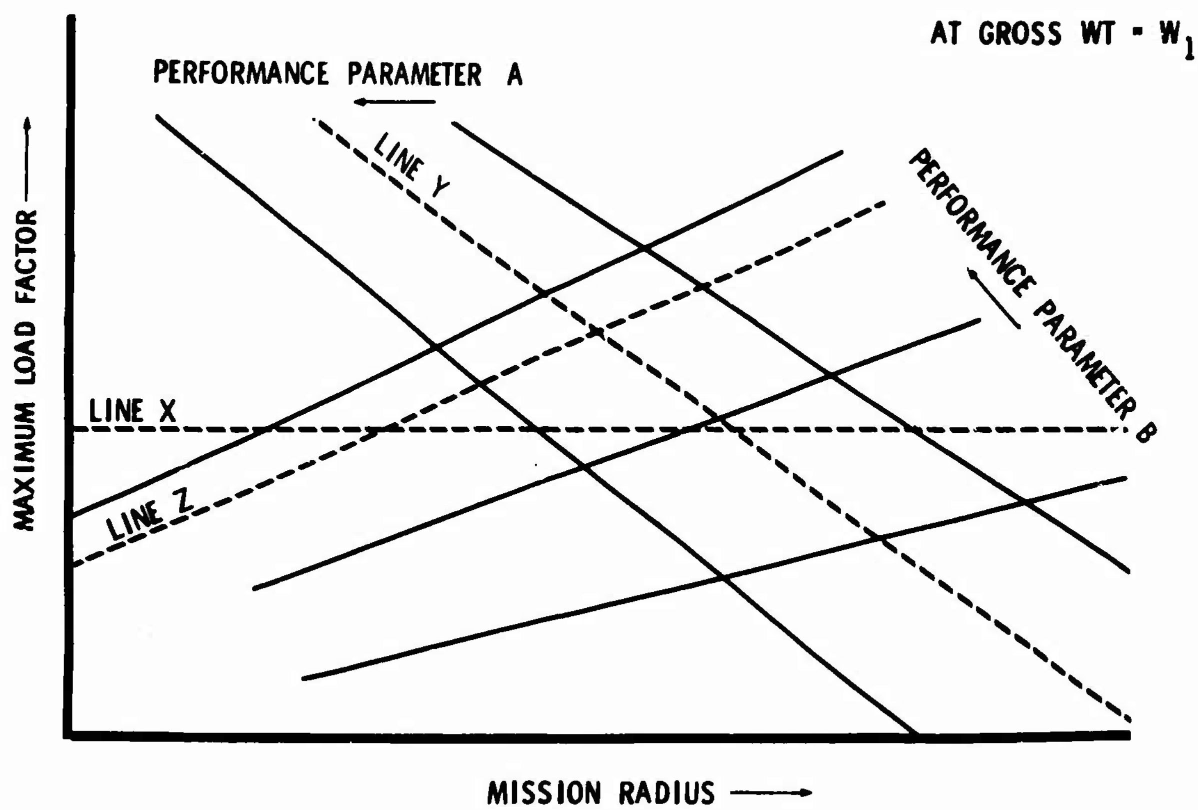

Fig.5 Technology plot – potential solution